UNCLASSIFIED

| AD NUMBER |
| --- |
| AD379233 |
| **CLASSIFICATION CHANGES** |
| TO: unclassified |
| FROM: confidential |
| **LIMITATION CHANGES** |
| TO: Approved for public release, distribution unlimited |
| FROM: Distribution authorized to U.S. Gov't. agencies and their contractors; Administrative/Operational Use; 30 JUN 1965. Other requests shall be referred to Federal Aviation Administration, Washington, DC. |
| **AUTHORITY** |
| FAA ltr, 10 Oct 1972; FAA ltr, 10 Oct 1972 |

THIS PAGE IS UNCLASSIFIED

# SECURITY MARKING

**The classified or limited status of this report applies to each page, unless otherwise marked.
Separate page printouts MUST be marked accordingly.**

Best Available Copy

Best
Available
Copy

NOTICES

When Government drawings, specifications, or other data are used for any purpose other than in connection with a definitely related Government procurement operation, the United States Government thereby incurs no responsibility nor any obligation whatsoever; and the fact that the Government may have formulated, furnished, or in any way supplied the said drawings, specifications, or other data, is not to be regarded by implication or otherwise as in any manner licensing the holder or any other person or corporation, or conveying any rights or permission to manufacture, use, or sell any patented invention that may in any way be related thereto.

Copies have been placed in the DDC collection. U.S. Government agencies may obtain copies from DDC. Other qualified DDC users may request, by submission of a DDC Form 1, through:

Director of Supersonic Transport Development
Federal Aviation Agency
Washington, D.C. 20553

Defense Documentation Center release to the Clearinghouse for Federal, Scientific, and Technical Information (CFSTI) and foreign announcement and dissemination are not authorized. The distribution of this report is limited because it contains technology identifiable with items excluded from export by the Department of State (U.S. Export Control Act of 1949 as amended).

# CONFIDENTIAL

Report No. PWA-2600
Date: 30 June 1965

Volume 3

# SUPERSONIC TRANSPORT AIRCRAFT ENGINE

# PHASE II-B DEVELOPMENT PROGRAM

# FINAL REPORT (U)

Prepared Under Contract FA-SS-65-18

Period Covered 1 January through 30 June 1965

PWA-E. H. Document Control
Eng. Sub-Control Station

JUN 25 1965

Number C63-5-C271-C



Approved by C W Bristol Jr

C. W. Bristol, Jr.
Project Engineer

Pratt & Whitney Aircraft DIVISION OF UNITED AIRCRAFT CORPORATION

EAST HARTFORD • CONNECTICUT

FAA SECURITY CONTROL
NO. 67571 A-18

COPY NO ____



# CONFIDENTIAL

PAGES NOT FILMED ARE BLANK.

# ITEM 2B

# STJ227 TURBOJET

ITEM 2B

# B. STJ227 TURBOJET ENGINE

## I. GENERAL DESCRIPTION

Refinements of climb path and airframe design have redefined the Supersonic Transport propulsion requirements. It was felt desirable to determine the effect of these changes on the relative capabilities of a fully augmented turbojet cycle and the duct-burning turbofan. Since a true comparison of the advantages of each type of powerplant can be made only with firm, consistent engine designs, a refinement of the turbojet engine configuration reported in Phase II-A was made.

The configuration and size chosen for the Phase II-B detailed study is a single-spool, three-bearing rotor engine having a nine stage compressor with a two-position inlet guide vane. The two stage turbine is an overhung design. The combustion systems selected for the study included a ram induction main burner and a short, fully augmented afterburner. The ejector-reverser configuration chosen is a sliding shroud, two-position, blow-in door ejector similar to that designed for the STF219 turbofan engine.

A review of current airframe-engine studies conducted by Boeing, Lockheed, and Pratt & Whitney Aircraft indicates that an engine having the following characteristics would be representative:

| | |
|---|---|
| Airflow (lb/sec) | 525 |
| Compressor Pressure Ratio | 9.26 |
| Turbine Inlet Temperature (°F) | 2000-2300 |
| Maximum Tailpipe Temperature (°F) | 3100 |
| Maximum Compressor Inlet Temperature (°F) | 525 |

The results of this design study show the configuration to be feasible, with many construction details similar to those now employed in the JT11D-20. The design study covered variations in number of compressor stages, and the comparison of 2300°F and 2000°F turbine inlet temperature turbine configurations. The conclusions to date are that the selected nine stage configuration is lighter and has more turbine temperature capability than a competitive eight stage design because of a lower operating rpm. Design studies have shown that the final turbine inlet temperature rating of 2300°F is a realistic development goal. The bases for these conclusions are the result of comprehensive analytical studies and extensive experimental testing of the JT4 High Temperature Turbine Development engine and the JT11D-20 at Mach 3 environmental conditions at temperatures in excess of 2200°F.

However, it should be noted that past experience on long life commercial engines has shown that only through service experience can the true life (TBO) be determined. Extended life and uprating of the engines (increased turbine inlet temperature) have been attained only by an extensive development program based on service experience.

# II. AERODYNAMIC TURBOJET STUDIES

## 1. COMPRESSOR

The compressor requirements for the STJ227 engine are listed below:

| | |
|---|---|
| Inlet Corrected Flow (lb/sec) | 525 |
| Pressure Ratio | 9.26:1 |
| Adiabatic Efficiency | 0.83 |
| Polytropic Efficiency | 0.875 |

Detailed design work has progressed on a nine stage compressor. The pertinent design parameters are listed below:

| Configuration | Number Stages | Inlet Specific Flow (lb/sec/ft$^2$) | RPM | Inlet Hub/Tip Ratio | Exit Mach No. | Avg $\Delta P/q$ | Avg Wheel Speed (ft/sec) |
|---|---|---|---|---|---|---|---|
| Constant Mean Diameter | 9 | 40 | 4800 | 0.477 | 0.38 | 0.41 | 864 |

The sea level static inlet specific flow of the compressor is 40 lb/sec/ft$^2$, which results in a specific flow of 42.3 at the transonic acceleration condition.

Figure 2B-1 is a plot of compressor efficiency versus specific flow resulting from JT11D-20 rig and engine testing. The sea level static and transonic acceleration operating requirements appear attainable. However, these requirements will dictate some additional development.

The use of a ram induction burner in the STJ227, with its higher allowable air inlet velocity, permits the use of a higher exit velocity from the compressor, with a resulting decrease in required wheel speed and a lighter weight compressor. Based on analyses of the burner, diffuser, and compressor combination, a design exit Mach number of 0.38 was chosen.

The design philosophy for this compressor utilizes a relatively low wheel speed, high flow coefficient (Cx/u) concept. This can be explained more fully by examination of Figures 2B-2 and 2B-3, which show the relationship between flow coefficient, work coefficient, and loading ($\Delta P/q$ and diffusion factor). For a known value of compressor work (enthalpy rise or pressure ratio) and aerodynamic loading

($\Delta P/q$ or diffusion factor), a compressor could be designed at either high wheel speed and low number of stages (low Cx/u) or low wheel speed with more stages (high Cx/u).

The selection between the two configurations can be made on the basis of weight and performance for each specific compressor operating envelope requirement. In general, compressor weight varies as the square of the wheel speed and directly with the number of stages. This would suggest the selection of the greater number of stages unless some other mechanical consideration controlled.

In addition to the weight consideration, attaining high cruise efficiency with minimum development is of considerable importance. High efficiency at reduced corrected speeds is easier to achieve with a fewer number of stages. At part speed, compressor front and rear stages are forced to operate with mismatched incidences. Reducing the number of stages helps to lower the range of incidence that these stages must accept.

Therefore, in addition to the nine stage low wheel speed compressor, a backup design utilizing eight stages with higher wheel speed is being analyzed. The pertinent design parameters for this compressor are listed below. Although final weight estimates have not been completed for this design, the preliminary weight estimates indicate the eight stage machine to be heavier.

| Configuration | Number Stages | Specific Flow (lb/sec/ft$^2$) | RPM | Inlet Hub/Tip Ratio | Exit Mach No. | Avg $\Delta P/q$ | Avg Wheel Speed (ft/sec) |
|---|---|---|---|---|---|---|---|
| Constant 3/4 diameter | 8 | 41.4 | 5240 | 0.485 | 0.382 | 0.382 | 1025 |

Flowpaths for the two designs are shown in Figures 2B-4 and 2B-5.

2. MAIN BURNER

The design objectives of a burner configuration include an efficient combustion process capable of delivering the specified temperature profile while having a minimum pressure loss. The ram induction burner shown in Figure 2B-6 was chosen as the one most capable of being developed to these objectives. An added advantage of this type burner system is the compatibility with the low wheel speed, low weight compressor design. Analytical studies of can-annular and annular burners of both conventional and ram induction type verified

the choice of this burner system. Recent testing under Phase II-A and II-B, plus other company-sponsored testing, has confirmed the study results.

Rig testing substantiating the design objectives has demonstrated temperature profiles of $1.1 \Delta TVR \quad \frac{\text{Max Temp Rise}}{\text{Avg Temp Rise}} = \frac{T_{t4max} - T_{t3}}{T_{t4avg} - T_{t3}}$ with 6-8% pressure loss at critical points of the SST flight path.

The ram induction burner utilizes a diffuser and turning vanes or scoops to prepare air for entering the combustor. The scoops partially diffuse the air and turn it efficiently into the burner. The velocity head carries the air into the burner, and thus no static pressure drop is required at the combustor wall. This design eliminates the need for part of the diffuser, thereby saving length and weight. It also reduces pressure drop, because the pressure loss associated with the extra length of the diffuser is eliminated. In addition, a large proportion of the pressure loss is used efficiently for mixing combustion gases, air, and fuel vapors.

Total system pressure losses are attributed to diffuser loss, scoop turning and dump loss, and a burning or momentum pressure loss. Current analytical estimates of pressure loss values are summarized as follows:

| | |
|---|---|
| Diffuser loss | $0.2\,(q_1 - q_2)$ psi |
| Scoop turning and dump loss | $1.3\,(q_2)$ psi |

where:

$q_1$ = compressor discharge velocity head psi

$q_2$ = velocity head in the plane of the scoop psi

These estimates of pressure loss predict rig cold flow test results. Burning or momentum pressure loss is directly calculated by a single heat addition equation and is a function of stream Mach number and burner temperature ratio. The total STJ227 burner system pressure loss estimates are shown in Figure 2B-7.

The diffuser configuration of the ram induction burner incorporates the burner forward section which forms a portion of the airflow paths. The small angle turns and relatively long length/diameter of each airflow path give a highly efficient diffusion process. Diffuser equivalent conical angles (ECA) fall between 5 and 6 degrees. The efficient stable diffuser operation assures smooth combustion and promotes the capability of generating a good turbine inlet temperature profile.

Compressor exit velocity profile shifts between sea level and altitude flight conditions. This flow shift generally weakens large ECA diffuser operation by causing local wall separation and translatory stall. Rig testing on ram induction burners indicates that the small equivalent conical angle diffuser in conjunction with this burner is insensitive to compressor profile shifts, resulting in negligible changes in burner exit temperature profile. Figures 2B-8 and 2B-9 illustrate compressor exit velocity profiles, and Figure 2B-10 depicts the resulting temperature profile variation.

An additional characteristic of the ram induction burner is that scoop walls are cooled by the high velocity air passing through them. Rig development tests to date have demonstrated the capability of scoops to protrude to the combustor centerline and still maintain structural integrity. This flexibility in scoop protrusion into the combustion gas path ensures the capability of air dilution of combustion products to any desired turbine inlet temperature profile. In addition, early rig development of the ram induction burner has demonstrated the ability to shift temperature profile from a cold-to-hot core. Figures 2B-11 and 2B-12 show the combustor exit temperature profile shift.

The selected combustor air distribution is illustrated in Figure 2B-13. Combustor wall cooling is accomplished utilizing both convective and film cooling. Pratt & Whitney Aircraft experience in combustor design ensures these methods of cooling to be satisfactory in d veloping a durable combustor configuration.

Combustor operation has been satisfactorily demonstrated over a wide fuel air ratio. Thermally smooth and stable operation from 0.00072 to 0.033 exceeds the expected operating range of 0.008 to 0.021.

Ignition will be provided by two high energy spark igniters located in the primary zone of the combustor. The annular configuration allows ease of flame propagation around the full circumference.

## 3. TURBINE

### a. Introduction

The turbojet preliminary design study centered on selection of turbine configuration and design criteria. Working within the basic findings of Phase II-A that a two stage turbine would best serve the requirements of the STJ227, studies were made of flow schedule, velocity ratio vs efficiency, and weight. The results of these studies are described in Section III-19. Experience has shown that many critical design problems do not become evident until a detailed design is undertaken. Therefore, a detail design was started before the final selection of size and flow schedule had been made.

The work accomplished on the turbojet engine has been less comprehensive than that described for STF219. Where possible, complementary or alternate approaches were taken in order to make maximum use of all design effort. For instance, the baffled blade has been used as the first approach with the three-hole blade as a backup. In general the conclusions reached in the STF219 report are applicable to the STJ227, except as influenced by the absolute size of the parts and the detailed blade shapes. The second stage of the turbofan sees lower temperatures than the turbojet because of the greater work required from the first stage of STF219.

The structural design requirement of 3000 hours time between overhauls (TBO) has been exceeded in all turbine parts. This was done without sacrificing the high efficiency necessary to meet the performance requirements of the STJ227. The cooling schemes used for the STJ227 are adaptations of proven designs used in the JT11D-20. The schemes used extend the calculated TBO beyond the 3000 hour goal for 2000°F turbine inlet temperature, and allow initially a limited life at 2300°F turbine inlet temperature. Continued improvements in cooling schemes and materials will be pursued to obtain a TBO of 3000 hours at the basic engine rating of 2300°F. The total cooling air requirements for the STJ227 turbojet engine (including the estimated seal and platform leakage rates) have been computed for the turbine airfoils (Figure 2B-14).

Figure 2B-15 shows the results of the heat balance study for the first stage vanes of the STJ227. In the case of convective cooling, the high degree of scrubbing and turbulence in the cooling air passages, which produces highly efficient cooling, also produces high pressure losses. The required supply pressure to the airfoils is shown as 95% of compressor discharge pressure.

This Curve is the locus of possible designs that are considered state-of-the-art in convective cooling techniques. The designs are based on experience gained from development of air cooled vanes for the JT11D-20 and operation of this engine at over 2200°F turbine inlet temperature at simulated Mach 3 conditions.

| Type of Cooling | Max $T_{t4}$ | % C/A |
|---|---|---|
| State-of-the-Art Convective | 2000 | 1.30 |
| | 2300 | > 4.0% |

These data indicate that a 3000 hour life for the first stage vane in a 2000°F turbine can be achieved with state-of-the-art convective cooling.

b. The Effect of Combustor Design On Turbine Life

Turbine durability is influenced to a great degree by combustor performance. To provide maximum life, the combustor must deliver a specified radial temperature profile and, at the same time, minimize circumferential temperature distortion in the form of local hot spots. An improper radial profile can cause excessive blade creep and excessive distress on turbine disk rims and case attachment areas.

Local hot spots reduce life by causing excessive local oxidation, erosion, and creep. The annular combustor proposed for the STJ227 engine has been designed to deliver an optimum radial temperature profile with minimum hot spots.

The basic burner design approach closely parallels that taken on the STF219. For a more detailed discussion refer to the Turbine Section of the STF219 report.

c. Cooled Turbine Design

(1) First Stage Vane

The cooling configurations for the STJ227 vanes and blades are discussed in Section III-5. These initial designs use the same basic schemes that were developed for the JT11D-20 which was designed for high Mach number, high turbine inlet temperature operation. Some modifications to these schemes have been made to obtain better temperature distributions consistent with the design requirements for the SST application.

Other configurations have been evaluated in the design of STF219 and are discussed in that section. In general, the studies are applicable to both engines except as qualified in the introduction.

The first vane configuration consists of a cooling supply tube installed inside a cast hollow airfoil (Figure 2B-16). Holes in the leading edge of the tube provide impingement cooling for the airfoil leading edge. Holes around the sides and trailing edge of the tube supply additional cooling air to obtain the desired flat temperature profile. One of the advantages of this scheme is the development flexibility allowed by the removable cooling tube. Both spanwise and chordwise temperature gradients can be controlled by providing holes in the cooling supply tube as requirements are determined from testing. Figure 2B-17 is a composite of results from static heat transfer rig testing of several cooling tube configurations of the same aerodynamic design. The significant point is the extremes of gradients obtained in these tests. This indicates that the cooling scheme can be tailored to produce approximately the desired temperature profile. An estimate of the metal temperatures that will be obtained in the vane design selected for the STJ227 is given in Figure 2B-18.

(2) First Stage Blade

The first blade configuration is an adaptation of the baffle scheme developed for the JT11D-20. The baffling arrangement is designed to provide several desirable features:

Correct metering of the flow to the three main areas (leading edge, mid-chord, and trailing edge)

Higher thermal efficiency by increasing cooling air velocity and vortex generation

Mixing of the three main flows at several locations spanwise to ensure chordwise equalization of cooling air temperature.

The baffle scheme is a second generation JT11D-20 blade cooling configuration. The original configuration consists of a single cavity, hollow blade with pins connecting the two sides. These pins provide both structural rigidity and some degree of cooling augmentation as fins. The pinned, standard, and diamond baffle blade flow characteristics are compared in Figure 2B-19. These photographs were taken during flow tests of a two-dimensional plastic model of the internal passage in the JT11D-20 blade (five times size). The lower plastic surface of the passage was coated with a solution of oil and graphite. The rig then was supplied with air so that the solution was washed by the air streams. The flow paths and vortices are clearly observed in the photographs.

It can be seen that the flow through the pinned blade is essentially straight with very little chordwise mixing. This system provides no regulation of cooling air to the three main areas of the blade. The standard baffle arrangement provides all the desirable features listed above, as demonstrated by the flow test. The third configuration is a system of baffles in a diamond arrangement with the gaps separating their ends sized to force increased mixing beyond that obtained with the standard baffle. During the flow test of this configuration it was clearly evident that a great amount of c' rdwise mixing was achieved. The first two configurations have been tested in heat transfer rigs and in development engines.

Because of its known characteristics of high convective efficiency and reasonable flow rate, the standard baffle scheme was selected as the 2000°F configuration for the STJ227. Some of the modifications in the STJ227 design included:

Blunting of the leading and trailing edges to obtain reduced heat flux and metal temperatures locally.

Moving the lower row of baffles toward the outer end to tailor the radial metal temperature profile to parallel that allowed by the stress distribution.

Varying the gap between the slanted baffles to provide better chordwise distribution of cooling air.

Figure 2B-18 shows the metal temperatures for the first stage blade.

The cooling schemes for the second stage blade are similar to the first stage. For the initial 2000°F requirements the cooling air will not be used for these airfoils. However, as the turbine inlet temperatures increase cooling air or improved materials may be required.

d. Advanced Cooling Techniques

As illustrated in Figure 2B-14, advanced cooling techniques will be required to demonstrate sufficient and efficient cooling at the turbine design temperature of 2300°F. All three basic cooling techniques (convection, film and transpiration) are being investigated to meet these requirements. The following discussion will serve to illustrate the potential of one of these methods, i.e., increased convective cooling efficiency.

More efficient convective cooling will require that the cooling air passages be designed such that the overall convective heat transfer coefficient will be maximum for a given cooling airflow rate and pressure drop.

The thermal efficiency of convective cooling is improved if the coolant flow passage through the airfoil is divided into many small passages having a large length-to-hydraulic diameter ratio. The coolant film coefficient is of the form:

$$h_c \sim \frac{K}{D_H} (R_e)^m (P_r)^n ,$$

where m and n $<$ 1.

For a given flow rate and constant thermodynamic properties of the coolant, this can be reduced to:

$$h_c \sim \frac{1}{D_H} \left( \frac{D_H}{A} \right)^m .$$

The hydraulic diameter is given by:

$$D_H = \frac{4A}{P} ,$$

$$\therefore \frac{D_H}{A} = 4/P.$$

And finally,

$$h_c \sim \frac{1}{D_H (P)^m} .$$

The coolant surface area is:

$$A_S = N \bullet P \bullet L.$$

The total cooling rate per unit temperature differential is:

$$h_c \bullet A_S \sim N \bullet (P)^{1-m} \bullet \left(\frac{L}{D_H}\right)$$

where:

$h_c$ - film coefficient

K - conductivity

$D_H$ - hydraulic diameter

$R_e$ - Reynolds number

P - Prandtl number

A - flow area of one passage

P - wetted perimeter of one passage

$A_S$ - total cooling surface area

N - total number of passages

L - length of one passage

This indicates that many small passages of large length-to-diameter ratio (L/D) produce the greatest total cooling ability. Therefore, at the higher coolant flow rate and heat capacity it is desirable to utilize cooling schemes that involve a large number of these passages. The present capability of obtaining these in a cast or forged airfoil is very limited. However, through the use of advanced techniques a highly sophisticated cooling passage system can be obtained as shown in Figure 2B-20.

It should be noted that the increased L/D is required only for higher cooling air flow rates. At low flow rates, the heat capacity of the cooling air would be exhausted before the air could exit from the blade and would cause the metal temperatures near the discharge to overheat. Thus, for low cooling air design, the L/D values for existing schemes such as the baffle blade provide the correct level of cooling for a desirable metal temperature profile.

An illustration of the potential gain in cooling efficiency of the advanced blade cooling scheme over a more conventional blade cooling scheme may be seen in the following table. Blade A is a development blade which has been thoroughly tested at gas temperatures over 2000°F.

A comparison of calculated and measured metal temperatures for this blade at a gas temperature of 1900°F and 0.52% cooling air is also shown in the table. The calculated blade total gas temperature of 2100°F corresponds to approximately 2200°F turbine inlet temperature (including burner profile) for the STJ227. Blade B is designed to pass approximately 2.3% cooling air at the same available pressure drop as for Blade A. To do this, the thermal efficiency of the cooling scheme must be sacrificed so that only a moderate amount of additional cooling is obtained. The advanced design, Blade C, utilizes a highly efficient cooling passage design so that the increased amount of coolant capacity is used to full advantage. It is seen that the calculated metal temperature is significantly decreased below that of Blade B. In fact, this level of cooling is thermally competitive with film cooling, but without the inherent necessity of aerodynamic losses caused by the angular discharge of air into the main stream.

In the particular design of Blade C (Figure 2B-20), the air was discharged out the trailing edge in a direction parallel to the main stream.

## AVERAGE METAL TEMPERATURE COMPARISON
### (Convectively Cooled Blades)

| Blade | Cooling Air psi | Cooling Air %* | Cooling Air (°F) | Gas Temperature | Average Midspan Temperature Calculated | Average Midspan Temperature Measured |
|---|---|---|---|---|---|---|
| A | 8 | 0.52 | 1100 | 1900 | 1710 | 1725 |
| A | 20 | 0.90 | 1100 | 2100 | 1800 | |
| B | 20 | 2.3 | 1100 | 2100 | 1704 | |
| C | 20 | 2.3 | 1100 | 2100 | 1498 | |

* Percent of turbine main stream air

Film and transpiration cooling, and the use of advanced construction for highly efficient convective cooling at the higher coolant flow rates, offer the potential to significantly improve the cooling effectiveness of turbine airfoils. The timely development of these techniques, as well as the development of improved materials and coatings, will provide the means of extending the life of the turbine beyond 3000 hours for turbine inlet temperatures of 2300°F.

c. Aerodynamic Design

The turbine is designed to provide an equivalent efficiency of 86% and is also limited to exit axial Mach numbers of 0.5 or less at the cruise flight condition. The Mach number limitation is established to provide good afterburner performance. Six turbines were studied to satisfy three cruise airflow schedules at two turbine inlet temperature levels. In addition, preliminary turbines were rough-designed for the two compressor configurations which were evaluated.

(1) Non-Free-Vortex Design

Testing at Pratt & Whitney Aircraft has indicated that there is a potential efficiency gain and weight reduction associated with a non-free-vortex turbine design. Figure 2B-21 shows the expected results. Because testing is incomplete, two turbines are being designed. One is a

free-vortex, high velocity ratio design and the other a non-free-vortex, low velocity ratio design at the same efficiency. The turbine described in this section is the high velocity ratio, free-vortex design.

f. Turbine Cooling Losses

The effects of cooling air on turbine efficiency were also investigated during Phase II-B. In addition to the thermodynamic effects on the overall engine cycle performance, there are some aerodynamic effects on the turbine performance. These effects are stated below:

- Cooling air injected into the turbine gas stream requires some portion of the main stream energy to accelerate the cooling air-to-main gas stream velocity levels. Turbine tests have been run indicating desirable methods of introducing this cooling air into the gas path with a minimum loss.

- Compromises in the airfoil design (mainly leading and trailing edge thicknesses) required to effectively cool and protect the airfoils from low cycle fatigue, creep and erosion add some losses to the turbine.

- Pumping losses in the blades also add to the turbine work requirements.

These losses were estimated to result in a total high turbine efficiency loss of approximately 1.5%. Even with these losses, the required turbine efficiency of 86% is maintained.

4. AFTERBURNER

Mission requirements of the STJ227 afterburner system include stable efficient augmentation at sea level, transonic, and cruise flight conditions. To satisfy these afterburner design goals, the engine configuration consists of a low pressure loss system capable of delivering high combustion efficiency over a wide range of fuel air ratio. Pratt & Whitney Aircraft experience with the JT11D-20 afterburner demonstrates the capability of afterburner development meeting these design goals.

The afterburner system includes a low pressure loss gas flowpath with a V-gutter flameholder system staggered aft and toward the engine centerline. Air leaving the turbine is axially oriented by turbine exit guide vanes and diffused to a velocity consistent with high performance combustion. The system cold flow pressure loss estimates are shown on Figure 2B-22.

Utilization of the staggered flameholder configuration reduces gas stream blockage, resulting in both low pressure loss and low flameholder lip velocity. The additional important result is favorable combustion characteristics. Development experience gained on the JT11D-20 has demonstrated the proper combination of flameholder geometry and low velocity necessary to give good stable flameholding without screech, maintain wide blowout limits, aid ignition, and support high combustion efficiency. Figure 2B-23 depicts combustion efficiency predictions for the STJ227 fuel air ratio range.

Fuel system requirements of a 15:1 turndown ratio dictate a dual-sprayring configuration. To meet these requirements a close coupled and a premix fuel system are joined. The primary zone incorporates sprayrings and flameholders in an integral assembly (close coupled). The premix system adds fuel between the flameholders, promoting a high degree of fuel coverage. The combination of these two systems enhances flame propagation by providing required flameholder turolence and mixing with good fuel ocverage.

Configuration studies of the low and high temperature engines indicate that cooling system changes are required to achieve design life as combustion duct cooling requirements differ between the low and high temperature engines. The low temperature engine utilizes turbine discharge air to maintain satisfactory combustion duct and nozzle flap temperatures. The high temperature engine, however, requires dilution of turbine discharge air with compressor air to maintain the same satisfactory temperature levels.

Significant engine changes occur in the required combustion length and fuel distribution. The high afterburner inlet gas temperature increases fuel vaporization rates, resulting in a 6 inch decrease of required combustion length. In addition, the fuel turndown ratio decreases to 12:1 resulting in new zoning requirements of the sprayrings and control.

Cruise point flow schedule shifts result in afterburner diameter changes to maintain satisfactory maximum combustion duct velocities. The following tabulations summarize the pertinent design parameters for a number of low and high temperature engine configurations studied.

## SST LOW TEMPERATURE AFTERBURNER AERODYNAMIC DESIGN FEATURES

| | Flow Schedules | | |
|---|---|---|---|
| | Low | Base | High |
| Turbine Exit Mach No. (Cruise) | 0.45 | 0.5 | 0.5 |
| Diffuser Area Ratio | 1.41 | 1.50 | 1.50 |
| Diffuser Equivalent Conical Angle (degrees) | 6.7 | 8.7 | 9.5 |
| A/B Duct Diameter (in.) | 59.7 | 62.0 | 65.3 |
| A/B Duct Ref Mach No. (Cruise) | 0.23 | 0.25 | 0.25 |

## SST HIGH TEMPERATURE AFTERBURNER AERODYNAMIC DESIGN FEATURES

| | Flow Schedules | | |
|---|---|---|---|
| | Low | Base | High |
| Turbine Exit Mach No. (Cruise) | 0.42 | 0.47 | 0.47 |
| Diffuser Area Ratio | 1.41 | 1.5 | 1.5 |
| Diffuser Equivalent Conical Angle (degrees) | 7 | 8.7 | 9.4 |
| A/B Duct Diameter (in.) | 57.6 | 58.4 | 60.2 |
| A/B Duct Ref Mach No. (Cruise) | 0.220 | 0.238 | 0.238 |

## 5. EJECTOR-REVERSER

### a. Introduction

The STJ227 exhaust nozzle system has been designed to meet the following exhaust system performance expressed in terms of thrust minus drag coefficient ($C_{FP}$) at typical power settings for given flight conditions.

| | | $C_{FP}$ (With 2% Secondary Airflow) | |
|---|---|---|---|
| Mach No. | Flight Condition | Low Temperature High Airflow Engine | High Temperature Low Airflow Engine |
| 2.7 | Cruise | 0.999 | 0.999 |
| 1.2 | Acceleration | 0.964 | 0.975 |
| 0.9 | Cruise to Alternate | 0.920 | 0.930 |
| 0 | SLTO | 0.980 | 0.980 |

$$C_{FP} = \frac{\text{(Gross Thrust of Primary Flow and Any Secondary and Tertiary Air) - (External Wave Drag and Ram Drag of Tertiary Air and all Internal Losses)}}{\text{Ideal Gross Thrust of Primary Flow}}$$

A systematic program has been followed through Phase II to select and design an operationally flexible, efficient exhaust nozzle system. The exhaust nozzle has been shown to be a sensitive component in the effectiveness of the propulsion unit and, therefore, must exhibit high performance with a minimum compromise in weight and mechanical complexity. In Phase I, all available exhaust systems were appraised in relation to the supersonic transport application. The field of exhaust systems has been reduced to the blow-in door ejector, based on considerations of internal and external performance, weight, mechanical complexity, and compatibility with the airframe of the Supersonic Transport.

Since the conception of the blow-in door ejector as an exhaust system, Pratt & Whitney Aircraft has accumulated thousands of hours of wind tunnel testing with afterburning turbojet and mixed flow turbofan ejector exhaust models. Included in this testing were comprehensive ejector model test programs for the TF30 and JT11D-20 engines. Much of what was learned in the development of the design of the STF219, TF30, and JT11D-20 ejector systems has been applied to the STJ227 turbojet blow-in door ejector-reverser design.

b. Nozzle Description

The blow-in door ejector is a variable geometry, self-actuated nozzle that aerodynamically adjusts to the correct expansion ratio as engine pressure ratio and flight Mach number vary. The blow-in door ejector and its operation are described in Pratt & Whitney Aircraft's Phase I Proposal for the Commercial Supersonic Transport Engine, Volume E-XI, Ejector-Reverser.

c. Performance Definitions

The parameter used to present the exhaust nozzle performance is the gross thrust minus drag coefficient, $C_{FP}$. This coefficient is defined as the sum of the gross thrusts from the primary stream and the secondary and tertiary streams, minus the sum of the external pressure or wave drag, the internal drag, and the ram drag of any tertiary air, all divided by the ideal gross thrust of the primary stream. The reverse thrust coefficient is defined as above except that the thrust and drag are of the same sign.

d. Fully Augmented Turbojet Exhaust System

(1) Nozzle Sizing

Optimum ejector diameters for a 525 lb/sec engine airflow were established based on aerodynamic and weight considerations. Nacelle sizing was accomplished by taking into account the necessary performance-weight trades. Complete gas expansion was assumed at cruise conditions, and the resultant diameter was reduced to give optimum nozzle performance. The optimized ejector diameters are as follows:

| Airflow | Low Turbine Temperature Engine Ejector Diameter (in.) | High Turbine Temperature Engine Ejector Diameter (in.) |
|---|---|---|
| Low | 72 | 70 |
| Base | 75 | 75 |
| High | 78 | 78 |

Nacelle diameters obtained by aerodynamic-weight sizing are given in all cases except for the low turbine temperature, low airflow engine where the minimum nacelle diameter was considered to be larger. In this case, a minimum nacelle diameter of 72 inches is required to maintain the minimum engine-nacelle clearance. Accessory and control packaging around the engine may require that the low airflow engines have slightly larger nacelle diameters than shown in the table above. However, preliminary airframe studies indicate that the ejector diameter may be sized to provide optimum area-ruling for the overall airframe-nacelle.

(2) Performance

An ejector-reverser system for the fully augmented STJ227 engine is shown in the engine cross section (Figure 2B-24). Since the major design effort has been directed toward the STF219 turbofan engine and nozzle system no specific test data have been obtained for the STJ227 full afterburning turbojet ejector configuration. However, many other afterburning turbojet type ejector nozzles have been tested. Performance for a nozzle closely simulating the STJ227 requirements is presented to demonstrate that with development effort the required performance levels can be attained.

(a) Typical Afterburning Turbojet Ejector

Figure 2B-25 is a schematic of a typical full afterburning turbojet ejector. Performance obtained from this model for the critical flight regions is presented in Figures 2B-26 and 2B-27, and is compared with the STJ227 nozzle performance goals. The design point for this ejector, was somewhat different than the STJ227, however, it still shows good performance at all critical flight conditions. Optimization of the ejector contours for application to the STJ227 would yield the desired performance levels.

(b) Exhaust Nozzles for a Partially Augmented Turbojet

Another exhaust nozzle system, wind tunnel-tested for a partially augmented version of the STJ227, was reported in the Phase II-A final report, Volume II. The results of this test program also substantiate the performance levels that can be obtained.

(2) Design Criteria

(a) Primary Nozzle

The location and type of the primary nozzle used in conjunction with an ejector nozzle can affect exhaust system performance.

The mechanically actuated iris or flap type convergent nozzles are normally used. A major function of the primary nozzle is to control the effective jet area which in turn controls engine operation.

A parameter associated with the primary nozzle that affects ejector performance is the spacing ratio. This is the ratio of the distance between the primary nozzle exit plane and the ejector exit plane, and the diameter of the primary jet area. Figures 2B-28 and 2B-29 show that spacing ratio can have a significant effect on ejector performance. These figures also indicate that the optimum spacing ratio may not be the same for all operating conditions.

All primary nozzle effects must be considered in the choice of type and placement of the best system for a given application.

(b) Secondary Flow

Secondary airflow (excess inlet air or boundary layer bleed air) normally originates in the engine inlet. In many installations it flows between the engine and the nacelle to purge the engine compartment, cool the engine, and reduce the pressure differential across the engine case. When the flow reaches the ejector it cools the ejector, cushions

the expansion of the primary stream, and helps eliminate over-expansion of the primary stream by opposing premature attachment to the ejector walls and partially filling the exit area. A minimum corrected secondary flow of 2 percent is required for cooling.

(c) Blow-in Doors

Tertiary airflow is the key to the blow-in door ejector feasibility as a large pressure ratio range exhaust system. The tertiary flow prevents the overexpansion that occurs during low pressure ratio operation of a fixed convergent-divergent nozzle.

Figure 2B-30 shows the effect of reducing tertiary airflow from the base configuration. In these tests the tertiary air was reduced by providing additional circumferential blockage in the tertiary air passage. The $C_{FP}$ between approximately 1.0 and the peak of the base curve in Figure 2B-30 is a result of pressure drag, tertiary induction drag, shocks in the ejector, angular loss of the ejector, and internal friction. Since there is a trade-off between the effect of tertiary air on internal performance and its induction drag, it is important that the blow-in doors are properly sized and shaped. When the doors are open, the flow contour (blow-in door boattail) should be as gentle as possible so that the tertiary flow undergoes a small momentum change. Angles of approximately 7° on the forward portion of the door and 15° on the rear portion work effectively in the transonic region, and the closed doors produce no drag at cruise conditions.

(d) Ejector Shroud

The internal contour of the ejector shroud has a definite effect on ejector performance. For good blow-in door open performance, the front (convergent) portion of the shroud must be at a small angle to ensure that the tertiary air does not suffer much of an axial momentum loss. The "throat" portion of the shroud must be large enough to allow the tertiary air to flow into the divergent section without restriction. For blow-in door closed operation, the "throat" section must be small enough to prevent performance distortion resulting from rapid over-expansion of the primary stream. Optimum nozzle performance levels will be achieved by the proper placement of the "throat" or minimum shroud section relative to the primary nozzle.

(e) Trailing Edge Flaps

The important design criteria concerning the trailing edge flaps are internal flow contour, external flow contour, weight, complexity, and sealing. The optimum flap configuration is a compromise between

these aerodynamic and mechanical design requirements. Consideration of damping and synchronization schemes ensures stable, efficient aerodynamic operation at all flight conditions.

(f) Reverser

The important parameters considered in the reverser design were reverse flow area, reverse discharge angle, spacing of reverser doors from the primary nozzle, and bleed flow (spoiled or unspoiled flow not reversed). Design of the fully augmented STJ227 reverser was based on the coannular reverser model data, as presented in the Phase II-A final report, Volume II. These data indicate that a satisfactory reverser can be incorporated into the ejector design to obtain the required reverse thrust levels with no appreciable engine suppression.

# III. MECHANICAL TURBOJET STUDIES

## 1. VARIABLE INLET GUIDE VANES AND CASE

### a. Description

The inlet case of the STJ227 has 28 vanes with movable trailing edge flaps. The fixed portion of the vanes provides support for the number 1 bearing compartment and stiffness for the engine mount. The movable flaps, comprising 3.4 inches of the 7.3-inch total chord at the O.D., are used to minimize the incidence variation of the air entering the first stage rotor blades. This provides more efficient compressor performance over the required operating range. The airfoil contours of the proper air direction into the first rotor (Figure 2B-31). The vanes contain provisions for anti-icing by the use of compressor bleed air.

### b. Construction, Loading, and Actuation

The inlet case weldment, consisting of inner and outer rings with integral vane feet, two-piece structural leading edge, mount ring and mount pads, and inner stiffening ring, is made of A-110 titanium. A general case construction is presented in Figure 2B-32. Welding the leading edges to machined integral feet on the inner and outer cases results in a minimum of welding and, hence, a minimum of weld distortion. The middle case flanges, which are necessary for containing structural vane loads, are enclosed and enlarged to form the integral front mount ring. Forged mount pads are welded into cutouts in the ring at the proper circumferential locations. (Refer to Section III-14, Engine Mounts.)

All leading edges are hollow to minimize weight and to provide a compressor discharge air passage through the vanes for anti-icing. This scheme is shown in Figure 2B-33. Six of the vanes are used for oil supply and scavenge, breather, and seal pressurizing lines to the number 1 bearing compartment. These tubes have brazed bosses at the inner and outer ends and are a replaceable part of the inlet case weldment, as in the JT11D-20. Assembly is accomplished by inserting

the tube inward until sufficient clearance with the inner ring flanges allows an induction coil braze of the inner or number 1 bearing boss to the tube. Similarly, the tube is then moved outward sufficiently by passing the already brazed bearing boss through a matching lightening hole. The outer boss, consisting of a combined mounting bracket and tube fitting for external plumbing, is then induction brazed. For storage or transportation, the outer bracket is bolted to the case.

However, for ease of assembly and proper seal alighment when assembling the inlet case and number 1 bearing housing, the outer boss is loosened while bolting the inner boss to the bearing housing and then tightened again for vibration damping and attachemtn of external plumbing.

The movable feature consists of titanium trailing edge flaps supported at the inner and outer ends by a steel pin-bolt extending the length of the flap. The pin is fixed relative to the rotating flap in such a manner that all motion occurs between steel-backed carbon bushings located at the inner and outer ends of the pin. The outer end of the flap is splined and bolted with the pin-bolt to a steel linkage arm that extends into a titanium synchronizing ring. The inside diameter of the synchronizing ring forms a track that serves two purposes: it allows the ring to be mounted on a series of steel-backed carbon bushings so that the ring rotates with no axial translation during actuation; and it serves as a groove for a steel slab-headed pin for attaching the link arms to the synchronizing ring.

The choice between rotation with translation and rotation without translation was based on a number of actuation schemes. With an expected flap rotation of 30° to 40° and a radius (linkage arm length - synchronizing ring to flap rotation centerline) of approximately 2.5 inches, the fore and aft ring translation would be approximately one-eighth of an inch. A direct tangential attachment of actuators to the synchronizing ring would then result in flexing of the actuator high pressure hydraulic lines. This is not considered practical or safe. A fixed actuator with ring translation would require a complicated and highly wear-susceptible bellcrank arrangement with the total actuator load in one sliding joint. Without ring translation, actuators may be attached directly to the synchronizing ring. This allows all sliding motion to be taken between the synchronizing ring pin and the vane flap linkage arm. The sliding

joint load is then the actuator load divided by the number of vanes driven. The joint stress is reduced and, hence, the tendency to wear. The joint, as proven by JT11D-20 testing, is simply a hardened steel, slab-headed pin through a slot in the steel linkage arm. The slab-head in the synchronizing ring track prevents rotation of the pin against the titanium synchronizing ring. Should excessive wear ever occur at this joint, only the easily replaced linkage arm or slab-headed pin need be replaced because no sliding motion occurs between other parts.

The estimated maximum possible air loads at various portions of the system, obtained by ratioing JT11D-20 loads directly as the maximum airflow and inversely as the number of vanes, are listed below:

| | STJ227 | JT11D-20 |
|---|---|---|
| Axial Air Load per Vane (lb) | 41.5 | 35.5 |
| Tangential Air Load per Vane (lb) | 94 | 77 |
| Center of Pressure to Center of Rotation, assumed as 1/2 of flap chord (in.) | 1.7 | 1.245 |
| Torque Due to Air Load (in. lb) | 161.5 | 101.9 |
| Total Shear Load on Pin-Bolt (lb) | 103 | 84.75 |
| Link Arm Length, center of rotation to center of synchronizing ring pin (in.) | 2.50 | 1.33 |
| Synchronizing Ring Pin to Link Arm Slot Load (lb) | 64.6 | 76.5 |
| Total Actuator Load Required (lb) (28 vanes STJ227, 20 vanes JT11D-20 due to air load) | 1810 | 1530 |

With the small diameter pin-bolt, the torque or load required because of friction between the pin-bolt and its bushings will conservatively be less than 10% of the airload.

Using actuators of the size used on the JT11D-20 (piston diameter = 1 inch) and the pressure available on the STJ227 (1500 psi), individual actuator load available is 1180 pounds. Therefore, two actuators of an existing design would be sufficient to drive the system. A micro-switch may be mounted on the case and actuated by a plate on the synchronizing ring to provide a cockpit indicator of vane position.

Thermal expansion differences occur during transients even though the case and ring are of the same material due to a difference in heat transfer coefficients. The inlet case walls and vanes are subject to a high mass and velocity of air, and heat or cool very rapidly, whereas the synchronizing ring is subject to a small mass and velocity of bypass air, and its temperature lags the inlet case by a considerably amount. The synchronizing ring support bushings are mounted to the inlet case by brackets designed to act as springs to relieve thermal transients between the synchronizing ring and the case and such that the ring remains concentric with the case, thus minimizing the variable flap position tolerance.

c. Anti-Icing

The inlet guide vanes require anti-icing whenever the compressor inlet static temperature drops below 40°F to prevent a degradation in engine performance due to ice formation. Several methods of de-icing (periodic ice removal) and anti-icing (continuous ice removal) are available, such as mechanical de-icers, chemical and coating anti-icing, and thermal de-icing and anti-ici .

The thermal anti-icing method lends itself to the STJ227 engine. This system involves flowing compressor bleed air through the fixed section of the inlet guide vane and film heating the variable section with air from the fixed section (Figure 2B-33). The flaps remain axial during the flight envelope where anti-icing is required, except during periods of loitering (i.e., awaiting landing clearance) when the engine is throttled back. The flow requirements will be approximately 1 percent of engine airflow based on JT3-D engine anti-icing experience and analytical calculations on the JT11D-20 anti-icing system.

The anti-icing system flow is controlled by two valves located in the supply lines from the bleed point. One is an on-off valve connected to a manually operated switch in the cockpit. The other is a throttling valve varying automatically as a function of compressor discharge temperature.

It may be possible to use a lower bleed pressure than compressor discharge as provided for by diffuser case bosses. A study must be made of any possible advantages in weight or performance using various compressor bleed points.

## 2. COMPRESSOR

### a. General Description

#### (1) Compressor Cases

The compressor outer wall is formed by stacking and bolting short case and stator assemblies together as shown in Figure 2B-34. The first, second, and third stage case and stator assemblies are made of titanium; the other six stages are made of Waspaloy (AMS 5706 and AMS

5544). Each case and stator assembly provides for an abradable shroud over the blade tips which allows a reduced blade tip clearance to be used, and a box structure to which the outer ends of the forged stators are brazed. The inner ends of the vanes have integral tabs that are riveted to a diaphragm. This diaphragm supports the interstage seal ring by rivets and spacers in radial slots to accommodate differential thermal growth. Both the diaphragm and seal ring are made of Hastelloy X (AMS 5754 and AMS 5536). Similar proven construction has been used in the JT11D-20 engine.

The compressor exit guide vanes are made of contour rolled strip stock inserted into pre-punched slots in the outer shroud ring, twisted, and inserted into pre-punched slots in the inner ring. The inner shroud ring is bolted to the adjoining diffuser case flanges; and the outer shroud ring is supported by the diffuser case outer wall, with provisions for thermal expansion and fabrication tolerances.

During engine starting, air is bled from the compressor at a location just aft of the fourth stage stator and discharged overboard through hydraulically actuated valves. Air is bled continuously from the plenum chamber just inside the bleed doors to pressurize the bearing compartment seals.

(2) Compressor Rotor

The Compressor rotor consists of nine stages with the first, second, and third stage disks and blades fabricated of titanium. These assemblies are supported by an Incoloy 901 (PWA 1003) front hub at the second stage disk, as the first stage disk is overhung. These disks are joined and positioned axially by cylindrical Incoloy 901 spacers that have integral knife-edge seals for interstage sealing, and are held together by a set of tiebolts. For thermal compatibility between the front hub and the number 1 bearing compartment, Incoloy 901 material was chosen for the front hub. The spacers are made of Incoloy 901 rather than titanium to reduce bolt load relaxation due to differential thermal expansion. A conical Incoloy 901 spacer with an integral knife-edge seal separates the third and fourth stage disks. The fourth through ninth stage disks and blades are made of Waspaloy (PWA1007), and are joined and positioned axially by cylindrical or conical Waspaloy spacers having integral knife-edge seals. Disks and spacers are held together by sets of tiebolts. The fourth stage disk is solid to prevent circulation of the eighth stage air around the second and third stage titanium disks. The fourth through eighth stage disks have rim cover plates that are used to duct eighth stage air outward along the rim. This reduces the disk

temperature gradient during ascent and descent transient flight conditions and obtains long low-cycle fatigue disk life. Disk thermal gradient is reduced further by using blades with short extended necks to isolate the disk rim from the primary flowpath. The ninth stage disk has conventional extended root blades with mechanical damping provisions as shown in Figure 2B-35.

A cone, positioned between the eighth and ninth stage disks, provides additional stiffness and support for the rotor. An additional set of tiebolts connects the inner flange of this cone, the rear hub, and the eighth and ninth stage disks. An auxiliary disk is overhung rearward from the outer tiebolts of the ninth stage disk, and is used to support a thrust balance control knife-edge seal. This disk provides close control of ninth stage seal clearance.

Each disk and blade assembly, spacer, and the front and rear hub are statically balanced. The rotor assembly is then dynamically balanced as a unit by adding weights to flanges provided on the first stage disk and the rear hub.

Throughout the rotor structure generous fillet radii, careful blending of intersecting surfaces, and the use of rounded or chamfered corners are provided to minimize stress concentration effects.

The thrust loading on the rotor is transmitted to the number 2 bearing from the rotor spacers through the cone between the eighth and ninth stage disks, and through the conical rear hub. Radial loads are transmitted to the number 2 bearing through the rear hub and to the number 1 bearing through the front hub.

The cold static radial clearances of all knife-edge seals in the compressor are minimized. These clearances are adequate to provide for maximum tolerance buildup and for maximum radial differential thermal and elastic growth during high performance steady-state operation. Light non-destructive rubbing may occur during transient thermals and overspeed operation.

Blade tip radial clearances are established to provide for maximum tolerance buildup, maximum radial and axial differential thermal growths during transient operation, and radial elastic blade and disk growths at normal high temperature operating speeds.

Axial gaps between the forward side of all rotating parts and adjacent stationary parts provide for the maximum total tolerance accumulation, the maximum axial differential thermal growth during transient opera-

tion, the maximum thrust bearing end play, and the maximum forward blade tip deflection resulting from surge gas loads. The forward stages of the compressor are spaced to allow for the abnormal deflections resulting from foreign object ingestion.

The axial gaps and the seal land lengths between the rear of the rotating parts and the adjacent stationary parts provide for the maximum tolerance accumulation, the maximum axial differential thermal growth during transient operation, the rearward deflection of the rear hub, the maximum disk rim deflection resulting from surge gas loads, the maximum forward vane deflection resulting from steady-state gas loads, and the maximum forward seal support diaphragm deflection resulting from steady-state static pressure loads.

b. Stress Summary

The compressor parts were sized so that the resulting stresses would be consistent with the minimum properties of the best available proven materials.

The highest ratio of actual stress to allowable stress occurs in the blade dovetails at maximum Mach number cruise. Typical dovetail attachment stresses are listed below.

DOVETAIL ATTACHMENT STRESSES

(Fifth-Stage)

| Yield Design | Blade Root | Disk Lug |
|---|---|---|
| Material | PWA 1007 | PWA 1007 |
| Speed (rpm) | 5376 | |
| Temperature (°F) | 975 | |
| Controlling Operating Condition | Max. Mach No. Cruise | |
| 0.2% Yield Strength (psi) | 110,000 | |
| Neck Tensile Stress (psi) | 18,000 | 21,850 |
| Neck Tensile Stress/0.2% Yield Strength | 0.16 | 0.20 |
| Tooth Bending Stress (psi) | 19,850 | 49,500 |
| Tooth Bending Stress/0.2% Yield Strength | 0.18 | 0.45 |
| Tooth Bearing Stress (psi) | 90,400 | 90,400 |
| Tooth Bearing Stress/0.2% Yield Strength | 0.82 | 0.82 |
| Tooth Shear Stress (psi) | 15,800 | |
| Tooth Shear Stress/0.2% Yield Strength | 0.14 | |
| Combined Stress (psi) | 37,850 | 71,350 |
| Combined Stress/0.2% Yield Strength | 0.34 | 0.65 |

The rotor disks were designed for standard commercial burst and yield margins, creep growth, and low cycle fatigue life. Typical stress data are shown below.

### TYPICAL DISK STRESS DATA

(Fifth-Stage)

#### Burst and Yield Design

| | |
|---|---|
| Speed (rpm) | 5476 |
| Average temperature (°F) | 1105 |
| Ultimate tensile strength (psi) | 160,000 |
| 0.2% Yield strength (psi) | 108,000 |
| Average tangential stress (psi) | 78,000 |
| Yield margin | 1.12 |
| Minimum allowable yield margin | 1.05 |
| Burst Margin | 1.375 |
| Minimum allowable burst margin | 1.20 |
| Maximum web radial stress (psi) | 52,000 |
| Allowable stress (%) | 59 |
| Maximum bolt circle radial stress, psi | 52,000 |
| Allowable stress (%) | 66 |
| Controlling operating condition | Max Mach No. Cruise |

#### Creep Design

| | |
|---|---|
| Speed (rpm) | 5376 |
| Average temperature (°F) | 1105 |
| Average tangential stress (psi) | 75,000 |
| 1.1 x 6000 hour 0.1% creep strength (psi) | 91,000 |
| Allowable stress (%) | 0.8 |
| Controlling operating condition | Max Mach No. Cruise |

#### Low Cycle Fatigue Design

| | |
|---|---|
| Speed (rpm) | 5088 |
| Temperature bore (°F) | 1125 |
| Temperature rim (°F) | 965 |
| Minimum bore low cycle fatigue life (cycles) | 28,000 |
| Minimum bolt circle low cycle fatigue life (cycles) | 100,000+ |
| Minimum rim low cycle fatigue life (cycles) | 8700 |
| Minimum allowable low cycle fatigue life (cycles) | 8000 |
| Controlling operating condition | Descent |

The first three stages of compressor blades utilize part-span shrouds for frequency control, vibration damping, and resistance to foreign object ingestion.

The shroud angles on all three stages are greater than the friction angle to prevent shroud locking and permit sliding friction damping. The shroud location has been selected based on JT11D-20 experience. Primary bladed disk resonance modes have been eliminated from the operating range of the engine by disk rim design and proper spacer placement. Bladed disk modes for the first and third stages are plotted in Figures 2B-36 and 2B-37.

The ninth-stage compressor blades utilize an extended root damping system to minimize blade stresses caused by excitation from the diffuser case struts. Energy is dissipated by a set of toggle weights bearing on the blade platforms under the action of centrifugal force. The sliding friction contact caused by the blade motion relative to the toggle weight produces the damping action. A similar configuration is used on the JT11D-20 turbine blades seen in Figure 2B-35.

Rotor tiebolts are designed to prevent loss of bolt preload and flange separation under the combination of highest aerodynamic loads, maneuver loads, and the loss of 10 percent of a single stage airfoils.

The maximum combined stress (direct tensile and centrifugal bending) does not exceed the 0.2 percent yield strength at steady-stage conditions. The maximum direct tensile stress at assembly will not exceed 70 percent of the 0.2 percent yield strength. The tiebolts are designed to transmit the maximum torque loads imposed without exceeding 57 percent of the 0.2 percent yield strength. Hydraulic bolt stretchers similar to those currently used on commercial Pratt & Whitney Aircraft engines will be used.

The compressor vanes deflect axially by gas loading and by the differential static pressure across the seal diaphragms. The resultant load produces a bending moment on the vane. The deflection and stresses are calculated by assuming that the vanes are variable-inertia twisted beams which deflect as a guided cantilever. Suitable vane chord lengths and thickness ratio limits were selected using extensive experience in vane analysis plus testing and development of gas turbine vanes.

c. Critical Speed

The stiff bearing critical speed of the rotor is designed at 30 percent above the maximum deteriorated rotor speed. This margin assures that the final rotor design will have adequate stiffness when coupled to the

engine case through the bearings and their support structures. Figures 2B-38 and 2B-39 illustrate the complete rotor stiff bearing mode shapes of the STJ227 and JT11D-20. The support structure of the Number 3 bearing is relatively flexible compared to the Number 1 and 2 bearing supports. This relatively stiff rotor and bearing-mount structure arrangement ensures that no critical rotor bending modes occur in the normal operating range of the engine. The effect of the soft Number 3 bearing is to depress the first rotor bending mode of engine vibration below the normal engine operating range, while maintaining second bending of the rotor above the operating range.

d. Aerodynamic Brake

The ninth-stage stator is used for aerodynamic braking. By rotating the stators to block a portion of the airflow, the engine windmilling speed can be limited to an acceptable rpm after an inflight shutdown.

The shafts of the Waspaloy vanes turn on Waspaloy-backed carbon bushings that are pressed into the outer case and assembled in the inner shroud with the vanes. With the inner shroud and the ninth-stage vane and case attached directly to the diffuser case flanges, and the shroud and the case approximately the same temperature, the loose fit between the shaft and the bearings will be sufficient to absorb small misalignments.

A lever arm is splined to the vane shaft at one end and pinned to a unison ring at the other. The pins are coated to reduce wear and are locked in place with a lockwasher. This design is essentially the same as the JT11D-20 two-position inlet guide vane.

Because the design selection criteria is deflection, the unison ring is made of Inconel (AMS 5665). The ring is rotated by two tangentially pneumatic actuators powered by an airframe supply system.

The actuators are fitted with spring-operated plunger locks to hold the vanes in their normal position. When it is necessary to rotate the vanes, gas pressure releases the plunger lock and permits gas to enter the piston chamber, actuating the vanes. Capability for inflight rotation back to the normal position will be provided.

e. Low Cycle Fatigue

(1) JT11D-20 Experience

Pratt & Whitney Aircraft has gained considerable experience during the development of the JT11D-20 on compressor disk temperature gradients resulting from rapid changes in compressor inlet conditions. During the development of the STJ227 engine, several techniques have been utilized to decrease the temperature gradients within the disks. One technique uses solid disks (no bore hole) to divide the bore into compartments. Air is bled from the highest stage pressure in that compartment into the bore and exhausted at the lowest stage pressure. This technique minimizes the bore air temperature and, hence, reduces the temperature gradients in the disks. The second technique heats the rim of the disk by bleeding bore air over the disk and through the spacer. The bore air keeps the rim at a higher temperature resulting in a lower disk temperature gradient. A natural extension of the rim heating system was the addition of coverplates to prevent seal air and rim heating air mixing. This results in a more effective heating of the rim. Figure 2B-40 shows the typical descent disk temperature characteristics of the JT11D-20. The first of the three illustrations shows in solid lines the temperature characteristics of the bore, rim, and compressor inlet during JT11D-20 descent.

As the compressor inlet temperature ($T_{t2}$) decreases, the rim temperature lags slightly but follows the general slope of $T_{t2}$. Large maximum bore-to-rim temperature gradients result because the bore does not follow the compressor inlet slope. This is caused by the heat capacity of the disks and the poorer heat transfer conditions at the bore relative to rim conditions where air scrubbing velocities are high. The second illustration shows the temperature characteristics bore-to-rim at the point of maximum bore-to-rim temperature difference. The solid line shows the characteristic temperature gradient for the JT11D-20. The solid line in the third illustration shows the effect on the maximum bore-to-rim temperature difference of the descent rate. A low descent rate will result in lower maximum bore-to-rim temperature gradients.

Because the descent normally starts after a long steady-state flight, the initial conditions of descent are well defined. The initial conditions of the ascent are not well defined and have a strong effect on the transient disk temperatures.

The ascent disk temperature characteristics of the JT11D-20 are shown in Figure 2B-41. As in the descent, the rim follows the compressor inlet temperature with a slight lag. The bore lags the compressor inlet temperature, but not to the degree of the descent. The solid line in the second illustration shows a typical JT11D-20 radial temperature distribution in the disk at the maximum bore-to-rim temperature difference. During ascent, the rim temperature is higher than the bore, which is the opposite of the descent. The solid line in the third illustration shows the effect of ascent rate on the disk maximum temperature difference for the JT11D-20.

(2) STJ227 Low Cycle Fatigue

Experience gained from the JT11D-20 has been applied to the STJ227 compressor design during Phase II-B. High Mach number engine testing, spin pit testing, and intensive analytical studies indicate that descent transient thermal gradients are the controlling factor in low cycle fatigue life. Figure 2B-42 illustrates the effect of thermal gradients on low cycle fatigue life. It can be seen from these curves that stress in the rim cycles ranges between 0 and 110,000 psi. This gives a steady stress of 55,000 psi with a cyclic stress of ± 55,000 psi. Referring to Pratt & Whitney Aircraft low cycle fatigue data, it is determined that this particular disk location will withstand 1000 cycles to failure. The method of arriving at the above stress calculations involves the use of an IBM disk plastic analysis program that can run the entire engine flight spectrum and calculate the stabilized stress and permanent growth distributions.

Initial temperature estimates indicate that descent transient gradients on the order of 250° $\Delta$T bore-to-rim exist in the disks as first designed. Using current Pratt & Whitney Aircraft low cycle fatigue design curves, it was determined that these disks were LCF limited at 1000 cycles. This was far short of the 8000 cycles desired. The first approach at increasing the LCF life was to reduce the disk average tangential stress levels. Figure 2B-43 indicates the amount of weight addition necessary to achieve the 8000 cycles. Obviously a weight penalty of a factor of two was too severe. Analytical studies determined that a $\Delta$T of 130° could be tolerated without a weight penalty and still achieve the required LCF cycles as seen in Figure 2B-44. A combination of two methods was proposed to achieve this 130° $\Delta$T gradient.

The first method used blades with short extended necks. This allowed the disk rim to be moved inward from the flowpath. The second method provided cover plates on both sides of the disk rims to duct airflow to rim sections. It was determined that with a combined approach the 130° $\Delta$T gradient could be achieved, and the 8000 LCF cycle requirement could be met.

The dashed lines in Figures 2B-40 and 2B-41 show the temperature characteristics for a disk with rim heating with cover plates. The rim temperatures will be higher at steady-state, but the maximum temperature gradient allows the design of a long-life compressor disk.

To further improve the rim LCF problem three material laboratory testing programs are being conducted to decrease disk rim stress concentration factors. The first method is a photoelastic analysis to determine the effect of elliptical radii on stress concentration. The second test is an electrical analog study of disk slot width-to-disk neck width, in combination with circular and elliptical radii, for the resultant effect on stress concentration. The third test consists of shot-peening low cycle fatigue specimens with stress concentrations ($K_T$) of 2 and 3 to determine the magnitude of fatigue life improvement. Figures 2B-43 and 2B-44 reflect the increase in LCF life when the stress concentration factor is reduced from $K_T$ of 3 to $K_T$ of 2.5. Photoelastic tests of a similar dovetail attachment on a JT11D-20 compressor disk indicate a $K_T$ range from 2.2 to 2.6.

Bolt holes and disk bores are seriously affected by thermal gradients. It has been determined analytically that the STJ227 disk bolt holes can meet the 8000 cycle requirement with the addition of approximately 10 percent of the disk weight at the bolt circle location. Pratt & Whitney Aircraft experience also indicates that while rim stress concentration factors are in the $K_T$=3 range, bolt holes more closely approach $K_T$=2. All gradients previously discussed involve the descent condition in which the disk rim is colder than the disk bore. The opposite case in which the rim is hotter than the bore results from ascent conditions and increases bore stresses. STJ227 studies indicate that bore stresses during ascent conditions do not limit LCF life.

Cover plates are conical rather than flat so that their stresses are relatively unaffected by radial thermal gradients.

The descent gradients are in excess of 130° $\Delta$T in the first, second, and third stage titanium disks. It is possible for a titanium alloy to absorb a thermal gradient approximately three times greater than can be absorbed by a nickel base alloy because of a lower coefficient of expansion and a lower modulus of elasticity.

3. DIFFUSER

a. General Description

The diffuser section of the STJ227 turbojet engine consists of a double wall diffusing section bolted to the eighth stage compressor shroud assembly at the forward end and to the inner and outer combustion ducts at the aft end. The diffuser case has integral supports for the annular burner, fuel nozzles, and spark igniters; flanged openings for clean bleed air; provisions for two tower shafts for starting and accessory drive; and bosses for routing lubricating oil and seal air lines through the struts.

The aerodynamic design and function of the diffuser flowpath are described in Section III-4 Combustion Chamber.

The diffuser section acts as a structural component supporting the number 2 and 3 bearings and their associated bearing compartment seals.

b. Physical Structure

The diffuser case assembly is a weldment fabricated from Inconel 718 nickel alloy (PWA-1009 and PWA-1033). This material selection is based on fabrication experience with JT11D-20 diffuser cases, which are similar restrained weldments. Experience has shown that initial fabrication is relatively straightforward, and that weld modifications are possible in the heat-treated condition. These characteristics are not attainable to the same degree in other high strength, high temperature alloys.

The diffuser inner and outer walls are fabricated from machined forged rings with sheet metal inserts and joined by eight equally spaced struts. The inner and outer walls are stiffened by two rings on each wall. One ring is located at the leading edge of the struts, and the other is near the trailing edge of the struts. This four-ring stiffening structure is similar to that used successfully on the JT11D-20 diffuser case.

The struts are fabricated using a machined airfoil standup butt-welded into the outer case with machined leading edge, machined trailing edge pieces, and two H-shaped stiffeners. The strut wall between the H-shaped stiffeners is sheet metal panels. The forward H-shaped stiffener is butt-welded to the leading edge forming a cavity to direct the bleed flow to the annular manifold. The rear stiffener connects the two rear stiffening rings and carries the bearing loads to the outer case.

The machined airfoil standup was used to obtain a better flow of stresses from the strut to the outer case and to eliminate the need for fillet welds.

The inner case stiffening rings, Figure 2B-45, are integral supports for the Number 2 and 3 bearings and carry the bearing loads, rotor unbalance loads, and thrust and vibratory loads. The structure formed by the stiffening rings and struts distributes these loads at the stress concentration points at each end of the struts where they join the case.

An annular collector manifold is provided around the inside of the inner case wall. It carries compressor discharge air bleed from slots in the leading edge of each strut to the bleed discharge struts used for cabin pressurization, anti-icing, and afterburner fuel turbopump air. The manifold also pressurizes the other struts to avoid collapsing loads on the strut sides, and removes the external pressure load on the sheet metal panels between the struts and stiffening rings.

Two of the struts house accessory drive towershafts. One shaft drives an airframe accessory drive gearbox and starter drive, and the other drives the engine accessory gearbox. These shafts are powered by bevel gear pinions that mesh with a bevel gear on the engine shaft.

The three bleed discharge struts have flanged openings on the outer case wall. The flanges are butt-welded to standups integral with the machined airfoil standups for the struts, as shown in Figure 2B-45.

The remaining struts route lines to the inside of the engine for lubricating, scavenge return, seal pressurizing air, seal vent to ambient, and vent lines for thrust balance. The lines pass through sleeves in the struts to avoid contamination of the clean air. The sleeves are butt-welded to the annular manifold at the inner end and welded to integral machined bosses at the outer end. The lines are brazed into flanged fittings at the outer end and bolted to the integral bosses in the case (Figure 2B-45). Each strut capable of accommodating three tubes, one of which may be a 1.750 inch diameter tube with heat shielding on 2.000 inch diameter without heat shielding.

Bosses for the 32 fuel nozzles are fabricated from machined standups with welded flanges. The standups are machined in eight segments of four standups each. The segments butt-weld directly to the machined airfoil section and the rear stiffening ring. The machined airfoil section extends to the rear surface of the nozzle segment. This allows the rear case outer flange to be joined by a circumferential machine butt-weld. This method of construction avoids the possibility of distortion caused by closely spacing many circular welds.

Alternative methods of construction are being investigated for various parts of the diffuser case to facilitate fabrication and repair. One area under study is a method of attaching the annular collector manifold inner wall using only butt-welding (Figure 2B-45).

The trailing edges of the struts have been cut away to accept the attaching brackets of the annular burner liner. One pin is used at each strut to hold the burner liner in place. V-groove blocks at the inner brackets on three struts serve as orientation guides and assist in alignment to engage the support pins.

Spark igniters for the burner are installed 90° apart in two bosses on the centerline of (and aft of) two of the fuel nozzles. The spark igniters are installed using a two-bolt flange in the boss.

c. Temperatures and Stress Levels

The diffuser case temperature will be approximately compressor discharge temperature (1100°F max), as a result of the high internal heat transfer coefficients.

The outer case is designed for internal pressure and is yield limited. Stress levels will not exceed the 0.2 percent yield strength for any pressure and temperature combination. The inner case aft of the rear stiffener is subjected to an external pressure and is elastic buckling limited. The actual buckling stresses in this area will not exceed 50 percent of the critical buckling stresses for any pressure differential and temperature combination in the operating conditions.

## 4. PRIMARY COMBUSTION CHAMBER

a. General Description

The primary combustion section includes the compressor discharge air diffuser, the inner and outer combustion chamber cases, the annular combustion chamber liners, the transition duct into the turbine, the fuel supply system, the ignition system, the fuel nozzle system, and various seals. Figure 2B-6 presents a schematic drawing of the combustion section.

The main burner in the STJ227 is an advanced-design, annular ram induction burner. This is a new concept in burner design, differing from the more conventional louvered burners in several important aspects. Airflow through holes in a conventional burner is induced by the static pressure differential between the inside and outside of the

can wall. This requires diffusion of compressor discharge air to a relatively high static pressure with its associated pressure loss in the diffusion process. In the ram induction burner the momentum of the compressed air is used to force air through turning scoops into the combustion chamber. This provides excellent mixing of the fuel vapors, burning gases, and dilution air due to the turbulence generated by the impinging air streams. The use of total pressure, instead of static pressure, to feed the burner requires less diffusion of the compressor discharge air. This results in a lower diffuser pressure loss and provides a higher velocity air for metal surface cooling. Because of the inherent excellent mixing characteristics in the ram induction concept, the burner is less sensitive to pressure variations over the outside surfaces of the liners. The forward section of the ram induction annular burner forms a portion of the diffuser section, resulting in a shorter, lightweight design.

The high velocity diffuser discharge air is forced into the combustion chamber through a series of scoops that inject the air in a direction perpendicular to the centerline of the burner. The scoop area is divided into the primary scoops and secondary scoop sections. Fuel is injected directly into the front of the chamber, where it mixes with air injected through swirlers concentric with the fuel nozzles. The primary scoops bring the mixture to the stoichiometric ratio, and the secondary scoops bring about good mixing and temperature dilution. The scoops are placed opposite each other on the inner and outer combustion chamber walls such that the incoming airstreams impinge on each other approximately at the centerline of the burner. The resulting turbulence provides excellent mixing and prevents the formation of hot streaks and rich or lean areas. This ensures more even temperature profiles at the turbine inlet.

Since the static pressure differential across the combustion chamber liner is less for a ram induction burner than for a conventional louvered design, the structural requirements for the former are correspondingly reduced. Furthermore, the "thrust" generated by the scoops produces a load opposite to the pressure loading which reduces the structural requirements still further. Good cooling by the high velocity airflow over the liner, supplemented by film cooling on the surfaces exposed to the hot gases, gives low metal temperatures. This will permit higher design stresses to be used and will extend the life of the parts because of less oxidation, corrosion, and fatigue.

The transition duct section of the combustion chamber consists of convectively cooled inner and outer converging ducts which accelerate the combustion gases into the turbine inlet. The static pressure drop produced by accelerating the gases into the turbine inlet induces cooling airflow from outside the combustion liner through the annulus formed by the double walls of the transition ducts and into the turbine. This will provide film cooling at the inlet guide vane root and tip. The air for this cooling is drawn off from the space between the combustion chamber case walls and the combustion chamber liners. Because of the smaller airflow requirements for convective cooling relative to film cooling, more of the available air can be used for the high degree of mixing necessary to reduce the gas temperature variation across the turbine inlet.

b. Compressor Air Diffuser

The diffuser was analyzed on a streamline basis to determine the effect of various irregularities on diffuser performance. The effect of wall curvature, various inlet profiles, mechanical tolerances or mechanical protuberances, thermal displacements, leakage and recirculation into the diffuser, and wake effects from the struts were all studied to determine factors causing local flow separation or instability. It is essential for good turbine inlet temperature profile that the diffuser deliver stable unseparated flow to the ram air scoops at all flight conditions. As a result of this study several important improvements in the diffuser flowpath became evident.

Recent combustion rig testing has shown that burner internal flow areas can be smaller in the combustion zone than in the mixing zone. Reduction of inlet size reduces the amount of divergence required from the diffuser inlet to the inlet of the ram air scoops. This minimizes wall curvature effects and provides a diffuser which is more conservative and less sensitive to disturbances that cau ow separation.

The design incorporates two central diffusers in the nose of the fairing and two outside main diffusers with reduced airflow. All diffuser passages were analyzed on a streamline basis to ensure reasonable levels of wall loading. The equivalent conical angle and equivalent diffusion area ratio were computed for each passage and are within the limits required for flow stability. Enough air is taken through the central diffusers to feed the air swirlers at the fuel nozzles, provide dome cooling, and feed the primary scoops. The outside diffusers flow the remainder of the air which is sufficient to supply the secondary scoops, main burner film cooling a transition duct convective cooling air,

and turbine cooling air. The central diffusers pick up air in a region where the boundary layer effects are negligible and diffuse it smoothly to a plenum region in front of the burner dome. The swirlers are then fed by static pressure differential between this plenum area and the interior of the combustor. A small amount of cooling air is injected through slots around the swirler cups. It is desirable to reduce to an absolute minimum the tendency for flow to separate from the main diffuser walls. For this reason, the remainder of the central diffuser airflow is reaccelerated through converging nozzles to a discharge point just ahead of the ram air scoops. At this point the reaccelerated air mixes with the main outside diffuser air and then feeds the primary and secondary scoops. Because it is converging, the portion of this air that passes through the converging nozzle is not likely to suffer flow separation. Furthermore, the rapidly moving air discharging from the nozzle aspirates the main outside diffuser air due to viscosity effects. Normally, the outside diffuser air and the nozzle air move at the same velocity (Mach 0.21). However, if the outside diffuser air tries to separate at some point downstream, the nozzle air will accelerate the slowly moving separated air and pull the flow back in toward the scoops. This tendency to collapse the separated layer, due to the "air ejector" effect, ensures a stable flow to the ram air scoops. Also, the effect of separation-inducing disturbances, such as leakage around the fuel nozzle supports, is much less critical due to the decreased sensitivity of this design.

c. Inner and Outer Combustion Chamber Case

A one-piece outer combustion chamber case extends from the rear flange of the diffuser case to the forward flange of the turbine case at the end of the outer transition duct. The outer case is sized for maximum burner pressure, which is 181 psia at Mach 0.7 sea level. The flanges were designed to the same stress and creep limitations as the STF219 burner cases (Ref. Section II-2, Main Burner). The case is fabricated from Inconel 718 (PWA 1033) sheet, which is rolled into a hoop and longitudinally welded with forged flanges welded on at each end.

The inner combustion chamber case extends from the diffuser case inner rear flange to the turbine inlet guide vane and seal support case. The case is fabricated from Inconel 718, and has forged flanges at each end and a forged section in the middle. The connecting sections consist of a cylinder and a cone fabricated from Inconel 718 sheet. These sections are of double wall construction, (smooth sheet outer wall and corrugated sheet inner wall) seal-welded at the nodal points.

This construction is used on the JT11D-20 and provides exceptional stiffness with very light weight. This case was designed with the same 1.3 buckling margin used in other Pratt & Whitney Aircraft commercial engines.

For hot section inspection, the outer combustion chamber case is unbolted and moved forward over the primary and secondary scoop sections. The radial bolts which hold the inner transition duct and turbine inlet guide vane retaining segments are then released, and the inner transition duct slides forward. Removing the turbine inlet guide vane outer retaining segments will permit removal and replacement of the first stage vanes.

d. Annular Combustion Chamber

The annular combustion chamber is divided axially into two sections. The forward section is the primary injection section. Air is injected into this section through a series of ram air scoops to provide a stoichiometric fuel/air mixture in the combustion chamber. There are three axially staggered primary scoops in both the inner and outer combustion chamber walls for each fuel nozzle. The scoops are formed of sheet metal weldments welded into the chamber walls. The aft section of the combustion chamber (secondary injection section) is located immediately behind the last row of primary injection scoops. The secondary scoops are shaped to ensure uniform mixing and dilution of the bearing primary gases and to furnish both a suitable temperature and temperature profile to the turbine. There are two axially staggered secondary scoops in each of the chamber walls for each fuel nozzle. Fabrication of the secondary section is similar to the primary section except for the inclusion of tab-ended cascade vanes that are punched and welded into the side of the secondary scoops. The scoop walls in both the primary and secondary sections are convectively cooled by air flowing through the scoops. Combustion chamber wall cooling is achieved by a boundary layer film produced by a series of louvers in the walls between the scoops. Fabrication of the scoops by casting, as mentioned in the earlier proposal, has undergone further study. It was found that, while costing less, a cast scoop would also have to be welded into the chamber walls as do the formed scoops. The low loading on each scoop along with convectively cooled walls dictated a thin scoop wall. This could not be achieved with a casting without a considerable amount of casting process refinement which would in turn increase costs.

A single sheet metal wall is placed between each combustion chamber scoop wall and its adjoining combustion chamber case. This wall serves to maintain the air velocity supplying the ram air scoops and to minimize radiation.

The short front section of the combustion chamber around the fuel nozzle discharge is a single wall construction. It is cooled both internally and externally. Internally, the wall is boundary layer film cooled with air issuing from holes located in the vicinity of the nozzle face. Externally, the wall is convectively cooled by initial air, which is being accelerated from the forward cavity into the ram air scoop feed passages. Development of the chamber front section has moved the fuel nozzle aft from earlier designs, thus eliminating the requirement of a double wall for transpiration cooling.

The swirl cup is installed by inserting it forward through machined bosses on the forward part of the combustor. A lockring is then fitted over the protruding forward part of the swirl cup, indexed with a tab on the machined boss, and tack-welded to the lip on the machined boss. These tack welds can be ground off at overhaul, allowing the swirl cups to be replaced if damaged. The lockring also has a machined radial groove to receive a flange on the fuel nozzle swirler. Air is admitted through slots in the lockring and fed out through holes in the swirl cups to provide film cooling along the inner edge of the forward combustor face.

The forward part of the combustor containing the swirl cups and air scoops is held in place by eight radial pins which are inserted through the diffuser case outer wall. This feature is shown in Figure 2B-46. Machined forgings welded to the front of the combustor protrude forward and mate with platforms machined into the cut-away trailing edges of the diffuser case struts. The inner platform has a V-shaped block that fits into a mating groove on the combustor fitting to assure proper alignment of the holes for the radial pins that pass through the outer platforms. The pins position the burner axially and concentrically with the diffuser discharge with the thrust load on the burner carried by the pins in shear. A radial clearance gap at each pin allows for thermal expansion and tolerance accumulation between the diffuser case and the combustor.

The materials chosen for the burner construction are the same as those used in the STF219 burner. The sheet metal is Hastelloy X (AMS 5536), the air guide swirlers are cast Stellite 31 (AMS 5382), the retaining pins are Waspaloy (PWA 1004), and the retaining inserts on the combustor nose are L-605 (AMS 5759).

e. Transition Duct

The rear or transition duct section of the combustion section is an annular passage that further mixes the burned combustion gases and leads them into the turbine section. The transition duct is composed of independent inner and outer shells, each of which is fabricated with double-wall construction. The walls are formed sheet metal Hastelloy X (AMS 5536) weldments. The wall is convectively cooled with air taken in at diffuser discharge pressure at the forward end of the wall and discharged at the aft end just forward of the turbine inlet guide vanes at turbine inlet pressure. The inner skin of each wall provides the structural support to withstand the pressure gradient. The outer skin of each wall provides a channel for convection cooling and also supports the inner wall. The transition duct walls are held rigidly with respect to each other at the forward end. The aft ends of the inner and outer skin of each wall are held concentric, but are free to allow for radial and axial differential thermal growth.

f. Fuel Injection System

The STJ227 uses 32 equally spaced dual-orifice fuel nozzles that are individually removable. To ensure concentricity between swirl air and injected fuel, the swirlers are made integral with the fuel nozzles. Each nozzle assembly slides radially inward through the diffuser case, where a flange on the swirler engages a positioning groove machined into the lockring of the swirl cup assembly. The nozzle assemblies are held by bolted flanges on the outer wall of the diffuser case. To reduce coking, the pressurizing valves are located outside the engine where the environment is relatively cool. The nozzle assemblies may be individually removed for inspection and replacement with the added advantage that the swirler may be examined at the same time.

g. Ignition System (Main Burner)

The STJ227 turbojet engine uses an electrical ignition system. This system consists of an exciter mounted on the outside of the engine and connected to two spark igniters that penetrate into the front section of the combustion chamber as shown in Figure 2B-47. The igniters are held in place by a two-bolt flange on the diffuser case, immediately behind the fuel nozzle bosses, and are easily removable for inspection or replacement. The spark igniter (single electrode annular gap plug) has been used in numerous Pratt & Whitney Aircraft engines such as the JT3, JT4, JT8D, etc. To seal against excessive air leakage, the igniter probe passes through a close-fit spherical sleeve which accommodates mechanical tolerances and thermal displacements.

h. Seals

The joints at the forward end of the inner and outer transition ducts and the joint at the aft end of the outer transition duct present a particularly difficult sealing problem. Even though there is a very small pressure gradient across these seals (2 to 5 psi), they are subjected to very high radial and axial thermal displacement. Furthermore, the seals must maintain good flexibility in a 1200° to 1800°F temperature environment.

Initial seal investigation involved the convoluted hoop seals used in the JT11D-20 aft transition duct seal. While these do a good job of sealing, a better seal design has been studied for use in the STJ227, similar to the seal used in the J75 afterburner nozzle. Hundreds of thousands of hours of flight experience with this seal has been accumulated, and the results indicate a very high degree of durability.

The seal consists of two thin metal sheets sandwiched together with alternating slots cut lengthwise to form a series of interlocking fingers. The strip is attached at one end and preloaded against an annular rub ring at the other. Radial displacements then load the strip as individual cantilever beams, instead of loading it as a hoop. This configuration is much more flexible than a hoop and results in greatly reduced fatigue cracking with the consequent improvement in durability. Variations of this sealing method are being proposed for the three joints previously mentioned. The slotted seals are Astroloy (PWA 1013) sheet metal plated in the rubbing area with molydisulfide (PWA 586-1) and the rub ring is L-605 (AMS 5759) cobalt alloy.

5. TURBINE

a. General Description

The STJ227 is a two stage, single spool, overhung turbine engine designed to operate at turbine inlet temperatures from 2000°F (initial rating) to 2300°F (basic rating) with good efficiency. Even at the 2000°F initial rating, the engine can be operated at 2300°F TIT for prototype enging ratings.

The first stage vanes are hollow castings supported at both ends and are convectively cooled. The first stage rotor has hollow, convectively cooled blades with extended roots and root damping. Coverplates on the disk rim carry the damper weights and seals and direct the cooling air through the

blade root. Second stage vanes are hollow castings, cantilevered from the outer turbine case. The second stage rotor will have a hollow, cooled blade for the basic rating of 2300°F turbine inlet temperature, but will be uncooled for initial operation at 2000°F.

b. Turbine Stationary Parts

The turbine rotor is enclosed in a forged one-piece case. The turbine case is bolted between the burner outer case and the front section of the turbine exhaust case. The vanes are air cooled and structurally supported at the platform attachment points to withstand airfoil gas loads. Fail-safe design has been used throughout to prevent a fractured vane or a shaft failure from producing a propagating failure.

The turbine case is a one-piece forging made from Waspaloy (PWA 1004) material. The case is flanged at the front and rear ends, and is double conical in shape. Internal grooves and flanges, integral with the case, provide support for the turbine stators.

First stage vanes are cast from the Pratt & Whitney Aircraft developed, directionally solidified nickel base alloy (PWA 664), a material with good high temperature strength and creep properties, excellent thermal shock resistance, and good elongation. This material has been successfully used in the first stage vanes of the JT11D-20. The inner support for the first stage vanes consists of an inner turbine nozzle vane case and an inner vane retaining plate. At the outer platform, the vane feet fit into slots in the front and rear retaining rings. These rings, which interlock with the forward turbine case flange, prevent axial and circumferential movement of the vanes. At the inner vane platform, the attachment restrains the vanes from axial and circumferential movement, but allows free radial movement between the vane and the inner support case. The radial looseness is required for thermal differentials between the inner and outer case. This prevents the number 3 bearing loads from being transmitted through the vanes.

The first stage vanes are cooled by internal impingement and convection by compressor discharge air entering the vanes through internal distribution tubes from a plenum between the outer platform and turbine case. The cooling air reaches the plenum by passing between the burner and the engine outer case and through holes in the vane retaining ring. The air distribution tubes inside the vane airfoils have a number of small holes that direct air to the inside of the leading edges and the sides of the airfoils. Small protrusions on the sides

of the tubes control the gap between the tubes and the airfoil inner walls to maintain effective convective cooling as the air flows around the tubes. The air exits through slots in the trailing edges of the airfoils.

The second stage turbine vanes are cast from IN 100 (PWA658) nickel base alloy. These vanes are cantilevered from the turbine case and are subjected to gas pressure loads on the inner vane supports as well as on the airfoils. The outer vane feet fit into circumferential grooves and axial slots in the turbine case, thus transferring gas loads from the vanes to the case. The lugs on the inner platforms on the second stage vanes engage mating lugs in an annular channel-shaped inner support fabricated from Waspaloy (PWA 687). This attachment is required for free radial movement between the vanes and the inner support. This method of attachment prohibits thermal differentials between the turbine case and the inner support from placing additional loading on the vanes. Knife-edged rings are riveted to the forward lip and bolted to the rear flange of the inner vane support. Compressor discharge air is taken from the plenum between the first stage vane outer platforms and the turbine case. The air flows through holes in the first stage vane foot rear retaining ring, between the turbine case and the first stage blade tip seal, and through grooves in the turbine case lugs and vane feet. It then enters the plenum formed by the turbine case and the second stage vane outer platforms. Air from the plenum flows into internal shells in the vanes, from which it passes through holes to impinge on the inside of the airfoil leading edges (Figure 2B-48). It then flows rearward along the inner sides of the vanes and is discharged through slots in the trailing edges of the airfoils.

The blade rub strips are fabricated from Hastelloy X (AMS 5754) to ensure high oxidation resistance and strength. The first stage blade rub strip is segmented because of thermal incompatibility with the turbine case. It is held in place by the first and second stage vanes and their supports, and by anti-torque lugs which mate with a circumferential groove and axial slots in the turbine case. The rub strip is cooled by high pressure compressor air traveling between the rub strip and the turbine case. In addition, a portion of this air is metered through grooves in the first stage vane rear support ring and blankets the inner surface.

The second stage blade rub strip is held in place by the second stage vanes and by a machined groove on the turbine exhaust case. Torque lugs on the seal engage lugs on a ring that is grooved to accept the second stage vane rear feet. This ring is flanged and bolted between the turbine case rear flange and the turbine exhaust case front flange.

All vanes are secured at both ends to prevent a fractured or burned vane from being carried downstream. In addition, the knife-edge seals have been designed to permit rearward travel of the rotor. In the event of shaft failure, the rotor shifts axially into the stator and dissipates the stored rotational energy within the engine. The resulting battering of the airfoils reduces their torque input to the rotor. Containment of broken blades is provided by the blade rub strips and the turbine case. The required turbine case thickness for adequate blade containment was determined by strength and toughness factors for the rub strip and case materials at their maximum operating temperatures.

Provision has been made for the inspection of the first stage vanes and the first stage turbine disk and blades without a complete disassembly of the engine. The burner case and combustion chamber outer liner slide forward, providing access to the bolted plates that retain both ends of the first stage vanes. Both plates and the combustion chamber inner liner slide forward. Individual vanes may then be removed and replaced. Access to first stage blades for inspection is accomplished by removing as many vanes as necessary.

c. Turbine Rotor

The turbine rotor is made up of the turbine integral hub and spacer, the first stage disk and blade assembly, the second stage disk and blade assembly, the interstage seal-spacer, the turbine rear diaphragm, and the second stage disk rear seal. The disk and blade assemblies are mounted on the integral hub and spacer by thirty-two 0.625 inch diameter tiebolts for the first stage, and by twenty-four 0.625 inch diameter tiebolts for the second stage. The second stage tiebolts also retain the turbine rear diaphragm and the rear seal to the second stage disk. The interstage seal-spacer is bolted between the disks and retained by the disk coverplate bolts at each end. The seal-spacer carries a series of knife-edge seals which align with seal lands on the second stage vane inner shroud, and stiffens the large diameter turbine disks to reduce vibration.

The first stage disk and blade assembly consists of the disk, 82 blades, and the first stage disk front and rear coverplates. The coverplates are retained on the disk by forty-one 0.250 inch diameter bolts and locate and retain the blades by means of an axially tight fit on the blade roots. The front coverplate serves as a feeder plate to distribute cooling air to the first stage blades, and has two sealing lands for the knife-edge seals which are mounted on the rear of the inner burner case and first stage vane support. This coverplate also

has damper weights that bear on the blade platforms to control blade vibration. The first stage rear coverplate absorbs the axial thrust from the blades, and carries a seal land for a knife-edge which is mounted on the second stage vane inner shroud. Both front and rear coverplates are slotted radially at the inside, and in addition, the front coverplate is slotted from the outside (between damper weights) to reduce stresses due to thermal gradients. The inside slots on the front plate terminate at the cooling air feed holes; the slots on the rear plate are aligned with the disk blade attachment lugs to prevent air leakage.

The second stage disk and blade assembly consists of the disk, 102 blades, and the second stage disk front and rear coverplates. The coverplates are retained by fifty-one 0.250 inch diameter tiebolts. As in the first stage, the coverplates retain the blades by an axial tight fit, and the front coverplate distributes cooling air to the blades.

d. Turbine Disks

Both turbine disks are machined from Astroloy (PWA 1013) forgings. Considered in the disk design are: creep life at the l ng-time engine operating conditions, low cycle fatigue life as affected by stresses due to thermal cycling, burst limit in relation to maximum engine overspeed, vibration characteristics, and stresses at the rim due to blade attachment. The rotor cooling system has been designed to avoid large temperature gradients across the disk webs.

e. Turbine Blades

The first and second stage turbine blades are cast from IN 100 (PWA 658), and have hollow cores which reduce weight and serve as a passage for cooling air. The blades are attached to the disks by multiple-serration fir-tree roots. This method of attachment is designed to withstand stresses and fatigue resulting from rotation, airloads, and aerodynamically excited vibration, and has proved reliable and efficient by many hours of high temperature operation in the JT11D-20 and other Pratt & Whitney Aircraft engines. The blades have integral cast platforms at the airfoil roots which, when assembled on the disks, form the inner surface of the flowpath, and serve as lands for knife-edge seals between the rotating and stationary turbine parts. The blades are of the extended root type, in which the root attachments are separated from the blade platforms and airfoils by extensions. These extensions isolate the root attachments and the disk rims from the hot turbine gases. The extended neck on the first stage blade also provides a more effective action of the vibration

damper weights which bear on the platforms. The extended neck on the second stage blades provides room for the cooling air passage inlet between the blade platform and the root attachment. The second stage blades carry interlocking tip shrouds which control vibration and provide an integral knife-edge seal to reduce leakage around the blade tips.

f. Rotor Cooling

High pressure air for cooling the turbine rotor is bled from compressor discharge through holes near the rear of the diffuser case inner wall into a large plenum at the front of the turbine first stage disk. In addition to supplying turbine cooling, the air in this plenum is used for engine thrust balance by loading the turbine rearward to offset the forward load generated by the compressor. The major requirement of the rotor cooling air is to cool the blades. However, suitable orifices and flowpaths are provided throughout the rotor to assure that the disks, etc., are cooled, that all compartments and chambers are ventilated, that inflow of hot turbine gases is prevented, and that uniform temperatures are maintained across the turbine disk. A second source of cooling air is obtained from the compressor just aft of the eighth stage stator. This air flows through holes in the compressor rotor spacer and into and through the turbine shaft cooling the second stage disk bore. This air exits through orifices in the turbine rear diaphragm.

Cooling air reaches the first stage blade airfoils through radial holes in the roots. There are three holes through each blade root which intersect the cast core passage in the blade root extension, as shown in Figure 2B-49. The root attachment slots in the disks are deeper than the blade roots and form a passage for the cooling air to reach the radial holes in the roots. Cooling air is distributed to the blade roots by holes in the first stage front coverplate. The rear coverplate prevents the loss of the cooling air to the rear. This cooling system minimizes heat input to the disk, and maintains low axial and radial thermal gradients in the attachments.

The first stage blade airfoils are convectively cooled by air passing radially through the cored passages and exiting at the blade tips. A series of baffles in the passages mix the flow and increase the velocity to improve convective cooling. This scheme is presented in Figure 2B-50. The baffles begin at midspan, and are positioned to provide high turbulence and mixing at the leading and trailing edges where most cooling is needed. The cooling air exits to the low pressure side of the airfoils at the tips.

The second stage blades are cooled in the same manner as the first stage, except the cooling air enters the blade through holes in the forward face of the extended neck. Because of the greater number of second stage blades and the narrower attachments, it is not practical to bring the cooling air through the root as on the first stage. To reach the second stage blades, the cooling air passes through holes in the first stage disk, through radial holes at the rear of the interstage seal-spacer, and is distributed to the blades by holes in the second stage disk front coverplate. The second stage blades are convectively cooled by air passing through the hollow core and utilize internal baffles similar to those of the first stage blades. The cooling air exits through the blade tips behind the knife-edge seal.

A number of small radial holes are provided near the front of the interstage seal-spacer to pressurize the chamber formed by the interstage seal-spacer and the forward part of the second stage vane inner shroud and seal support. Air entering this chamber leaks out past the knife-edge seals at the first stage disk rear coverplate and the first stage blade rear platform. Cooling air also passes from the chamber between the disk through holes in the second stage disk to cool the rear face of the second stage disk. The cooling air at the rear of the second stage disk leaks past a knife-edge seal at the rear of the second stage blade platforms, and past the double knife-edge seal mounted at the disk tiebolt circle. The outer seal at the blade and the seal lands for the inner rear seal are supported from the inner turbine exhaust case.

g. Turbine Shafts

The turbine shaft is coupled to the interstage turbine hub at the number 3 bearing, and to the rear hub of the compressor at the number 2 bearing. In addition to transmitting torque and axial load, the hollow center serves as a flowpath carrying cooling air from the compressor to the turbine. The number 3 bearing provides support for the turbine rotor through the interstage turbine hub.

The rear flange of the interstage turbine hub is bolted to the front face of the second stage turbine disk. Just forward of this flange the hub branches into two sections. One cylindrical section extends forward to the front flange, which is bolted to the rear face of the first stage disk. The other section tapers inward, becomes cylindrical just aft of the first stage disk bore, and terminates just forward of the disk. The inner cylindrical portion, which passes through the disk bore, is internally splined to drive the turbine shaft. The hub has a shoulder aft of the spline that acts as a bearing surface for the coupling nut

that screws onto the end of the turbine shaft. The shaft has an external spline at the rear end to match the hub spline. From this spline the shaft extends forward, tapering slightly inward, to couple with the rear compressor hub just forward of the number 2 bearing. The shaft is splined at the forward end to drive the compressor hub. The hub and shaft are attached by a threaded coupling that screws into the turbine shaft and pulls up against a shoulder on the hub. The hub and shaft are locked together by a sliding spline spring-loaded mechanism to prevent loosening. The turbine shaft and hub form a double shaft that passes through the number 2 bearing. This feature ensures that the shaft cannot be twisted off by a main thrust bearing failure, thus preventing destructive turbine overspeed. The threaded coupling has an internal spline at the rear to allow for disassembly of the engine from the rear.

h. Turbine Rotor Balancing

Before assembly, the turbine shaft is dynamically balanced by removing material from the rear of the shoulder that acts as a stop for the rear compressor hub, and from the front of the shoulder that acts as a stop for the number 3 bearing assembly. The first stage disk and blade assembly is statically balanced by pairing the blades by the moment weight method during assembly to permit service replacement of blades in pairs without rebalancing, by mounting the front and rear coverplates with heavy sides opposite, and by inserting plugs in holes provided between tiebolts in the disks. The plugs are trapped at assembly by adjacent flanges. The second stage disk and blade assembly is balanced in the same manner as the first stage.

The turbine rotor is then assembled with the number 3 bearing compartment and support, the combustion chamber inner and outer cases forward to the diffuser case rear flanges, the turbine case first and second stage turbine stators, and all associated hardware. The entire assembly is then balanced by using nuts (classed by weight) on the first and second stage turbine tiebolts.

i. Stress Summary

Turbine component stresses were determined by an IBM program that determined the stress redistribution resulting from short-time local yielding and long-term creep. In addition, methods evolved through an extensive program of material and stress evaluation were used to predict burst and yield speeds, transient yield growth, creep characteristics, and low cycle fatigue characteristics of the turbine components. On the basis of these predictions, the turbine was designed to meet the requirements of the engine.

A definition of the factors used for the design of the first and second stage disks is shown below:

- Burst margin - overspeed above the maximum operating speed to cause fracture.
- Yield margin - overspeed above the maximum normal operating speed to cause general yielding.
- Creep growth not to exceed 0.1% - permanent growth due to a combination of temperature and stress over the life of the engine not to exceed 0.1%.
- Low cycle fatigue life - number of cycles (ascent, steady-state, and descent being one cycle) without producing cracks.

Allowable stress levels to provide adequate burst, yield, and creep capability and low cycle fatigue life were determined using experience gained from development of the JT11D-20 engine.

Waspaloy (PWA 1016) and Astroloy (PWA 1013) were considered for disk material. For both stages Waspaloy disks were creep-limited, whereas Astroloy disks were burst-limited. Since Astroloy disks are considerably lighter for each stage (46% first stage disk and 20% second stage disk) this material has been chosen.

The turbine disks were analyzed for the high flow cruise condition at Mach 2.7, 65,000 feet. This condition presents both the maximum speed and the most critical high temperature environment.

The following is a stress summary for the first and second stage turbine disks.

STRESS SUMMARY
DISK

| | First Stage | Second Stage |
|---|---|---|
| Disk Material | Astroloy | Astroloy |
| Average temperature (°F) | 1230 | 1215 |
| Maximum gradient (°F) | 35 | 130 |

| | | |
|---|---|---|
| Average tangential stress (psi) | 73,300 | 73,600 |
| Yield margin (%) | 17 | 16 |
| Burst margin (%) | 30 | 30 |
| Low cycle fatigue life (cycles) | | |
| Bore | 60,000 | 20,000 |
| Main bolt circle | 32,000 | 45,000 |
| Rim spacer bolt circle | 60,000 | 70,000 |
| Coverplate bolt circle | 80,000 | |
| Rim | 40,000 | 80,000 |

Blade-to-disk fir-tree attachments are similar to those that have many hours of successful running at similar temperatures and stress levels on the JT11D-20. The allowable stress levels have been derived from both JT11D-20 and other applicable experience. The following is a stress summary of a typical root attachment.

STRESS SUMMARY
ROOT ATTACHMENT

Disk material Astroloy

Attachment temperature 1255°F

Blade material IN 100

| BLADE STRESSES | Stress (psi) | Allowable Stress (psi) | Stress Factor Actual Stress / Allowable Stress |
|---|---|---|---|
| P/A (1st Neck) | 46,500 | 51,600 (65% 6000 hr. S.R.) | 0.90 |
| Mc/I (1st Tooth) | 29,650 | 33,800 (40% 6000 hr. S.R.) | 0.88 |
| Shear (1st Tooth) | 25,100 | 25,380 (30% 6000 hr. S.R.) | 0.99 |
| Combined (1st Neck) | 76,150 | 85,400 (105% 6000 hr. S.R.) | 0.90 |
| Bearing (1st Tooth) | 63,400 | 75,900 (80% x 0.2% yield) | 0.84 |
| DISK STRESSES | | | |
| P/A (1st Neck) | 43,500 | 66,200 (60% 6000 hr. S.R.) | 0.66 |
| Mc/I (1st Tooth) | 37,400 | 41,700 (40% 6000 hr. S.R.) | 0.90 |
| Shear (1st Tooth) | 24, [illegible] | 30,600 (30% 6000 hr. S.R.) | 0.79 |
| Combined (1st Neck) | 80, [illegible] | 107,000 (100% 6000 hr. S.R.) | 0.76 |
| Bearing (1st Tooth) | 61,500 | 100,000 (80% x 0.2% yield) | 0.62 |

The shafts and hub are designed to withstand the stress produced by a 10% blade loss load. Evaluations for torsional shear stress, creep, and buckling stress indicate that they are critical speed limited. The shafts and hubs for the STJ227 provide for a stiff bearing critical speed margin of 30%. This assures that the rotors, when coupled to the bearing support structure and cases, will not produce any bending modes in the operating range of the engine. Waspaloy (PWA 1007) is being used for these parts.

j. Turbine Vibration

The first stage turbine incorporates damped, extended root blades because of high blade buffeting excitation from the burner. Damping is accomplished in a manner similar to the JT11D-20 first stage by the use of centrifugally-forced damper weights. The damper weights contact the forward blade platforms and a land on the coverplate. Each weight can adjust radially, but is retained by a rivet to the forward disk coverplate. Relative motion between the blade platforms and coverplate is provided by the flexible blade root extension. In the fundamental blade vibration modes, friction forces caused by the damper weights rubbing on the contact surfaces damp the motion and reduce stresses. Damping effectiveness is achieved by properly sizing the damper weights for optimum friction force. Figure 2B-51 shows the maximum blade airfoil vibrational stress versus normal damper force for the JT11D-20 engine. The STJ227 design will provide similar damping, which will minimize blade vibrational stress levels.

The first stage blade and disk assembly has been designed to eliminate disk rim resonance in the running range. Figure 2B-52 shows the bladed-disk coupled mode. A 4-Nodal diameter mode lies in the running range, but this mode will have very little excitation.

The second stage blades are tip-shrouded to increase blade stiffness and blade resonance above 2E at speed. Blade stresses are reduced by positioning the shroud angle parallel to the blade bending motion in the fundamental vibration mode. Maximum damping is thus assured in the modes where maximum stresses are expected, with adequate damping in other modes. A disk diaphragm spacer was found necessary to stiffen the disk to provide the 2E margin in the bladed-disk coupled vibration mode.

k. Blade and Vane Structural Considerations

(1) Geometric Limitations

The aerodynamic requirements of the turbine primarily determine the size and shape of the airfoils as well as the approximate stress levels. The cooling scheme used must be capable of doing its job within the space allowed. Some compromises in the shape of the airfoils are acceptable, such as increasing the leading edge radii. This significantly reduces the outside film coefficients and reduces the temperature at the leading edges without seriously penalizing the turbine design point efficiency. To a lesser degree, the trailing edge design may be compromised by increasing the thickness to extend the cooling passage further. Some of the loss is regained by discharging cooling air out the trailing edge to energize the wake.

(2) Design Criteria

Controlling design criteria of turbine vanes and blades include creep and stress rupture, thermal fatigue, and erosion-corrosion. These three criteria are sensitive to metal temperature; therefore, the life of an airfoil can be controlled by designing for acceptable temperature levels.

In the case of creep and stress rupture the acceptable levels are based on long time steady-state operation where progressive creep will eventually reach levels at which the airfoils cease to be acceptable. In vanes the primary creep problem is associated with bowing of the trailing edges. Creep causes the blades to stretch until the tips rub against the case or until stress rupture cracks start. Either of these occurrences may end the useful life of a blade.

Thermal fatigue is associated primarily with transient stresses and strains caused by the high thermal gradients in the airfoil during engine transients. If the strains significantly exceed the yield limit of the material, the airfoils will be damaged during each thermal cycle. Careful design is required to obtain adequate cyclical life for the required time between overhauls.

Erosion and corrosion are processes by which the surface of the airfoils erode and thermally-activated corrosive action takes place at the inter-granular boundaries of the airfoil material. Coating developed for elevated operating temperatures will protect the base

material. The coating itself, however, may eventually erode to the point that the gas can reach the base material. Therefore, one design requirement for the airfoils is that a surface temperature be maintained that will permit the coating to remain intact for the specified time between overhaul (TBO).

## 6. TURBINE EXHAUST

### a. General Description

Because of unique design, the turbine exhaust section of the STJ227 is lighter than the JT11D-20. By using overhung turbine disks, the need for major structural support at the rear of the turbine rotor has been eliminated. The turbine exit guide vanes do not accommodate the oil, air, and breather lines and do not carry bearing loads to the outer case. The construction and assembly of the section were, therefore, simplified.

The airflow leaving the turbine is diffused through the exhaust section which has 4° inner and outer case cone angles. Swirl is removed by 16 exit guide vanes before the flow passes into the afterburner. The exit guide vanes are spaced far enough rearward to minimize second stage turbine blade excitation, as shown in Figures 2B-53 and 2B-54.

### b. Physical Structure

The exhaust flowpath is formed by the inner and outer turbine exhaust cases, which are machined from Waspaloy (PWA 687) forgings. Waspaloy cases on the JT11D-20 have demonstrated satisfactory performance at comparable temperatures.

The inner case is flanged at both ends. A knife-edge seal and a diaphragm-supported knife-edge seal land are riveted to and supported by the front flange. This seal structure controls cooling airflow at the rear face of the second-stage turbine disk. The rear flange supports the truncated tailcone and a diaphragm that closes the cone's forward end.

The tailcone forms the diffuser inner flowpath and is truncated just forward of the afterburner spraybars. The inner diaphragm prevents hot exhaust gas recirculation inside the inner case. Two circumferential stiffener rings are an integral part of the inner exhaust case and serve as the supporting structure for the exit guide vanes.

The outer case supports and positions the second-stage turbine blade rub strip at its forward end and the afterburner-ejector system at its rear flange. The rear engine mount is located approximately midway between the front and rear flanges. Maneuver loads imposed by the engine mount and the ejector system dictate case material thickness. The case is buckling limited and is designed to give an elastic buckling margin of 2.

The exit guide vanes are cast from Inconel 713 (AMS 5391) with an integral inner foot, and are cored for minimum weight. Sheet metal construction was considered for the vanes because of its light weight, but experience with both types of vane on the JT11D-20 dictated a choice of castings for long life and trouble-free operation.

The vanes extend through cutouts in the inner turbine exhaust case and are bolted to the stiffener rings (Figure 2B-55). A radial pin machined from Waspaloy (PWA 687) is extended through the outer turbine exhaust case and mount ring into a socket in the tip of each exit guide vane to form the outer vane support. The vanes are individually replaceable so that damage to one or more vanes does not necessitate the installation of a completely new inner exhaust section structure.

The pinned outer vane attachment allows radial vane growth during transient thermal conditions independently of the outer exhaust case. This construction eliminates a stress problem which would result from attaching the vanes rigidly to both inner and outer cases. The absence of a bearing behind the turbine makes the sliding pin attachment possible. Vane loads are primarily due to the gas load resulting from turbine swirl. Stresses are limited to the 6000 hour stress rupture strength of the vane material.

7. BEARINGS AND SEALS

a. General Description

(1) Bearings

The number 1 bearing compartment is supported by a box structure formed at the inner end of the inlet guide vanes. An oil scavenge pump in the compartment is driven from the front hub of the compressor rotor. The number 1 roller bearing is preloaded by means of an elliptical outer race to prevent roller skidding. The bearing material is PWA 724 or PWA 725 steel melted by consumable electrode vacuum procedure, stabilized for 600°F.

The bearing compartment is oil-jet cooled and lubricated. The main shaft seal is an assembly of carbon ring seals, which are loaded by fourth stage compressor discharge air, backed up by labyrinth seals. This backup system prevents leakage of oil vapor into the main engine flowpath, which otherwise could cause contamination of the clean cabin air supply system tapped off the main stream at the diffuser case. The seal schematic is shown in Figure 2B-56. The material for the knife-edge section of the labyrinth seals is Hastelloy W (AMS 5755) and the seal land material is stainless steel (AMS 5613).

Fourth stage air is obtained from transfer tubes routed through inlet guide vanes. In addition, oil supply, return, and breather lines are routed through other inlet guide vanes.

The number 2 bearing compartment is located inside the diffuser flowpath wall and is supported by flanges attached to the inner diffuser case. The number 2 bearing is a ball thrust bearing from which the load is transmitted through the struts to the outer diffuser case. The bearing material is PWA 725.

This bearing compartment is oil-jet cooled and lubricated. A carbon face main shaft seal pressure loaded by fourth stage compressor discharge air and backed up by labyrinth seals prevents oil vapor loss to the inner burner case region. The seal schematic is shown in Figure 2B-57. Hastelloy W (AMS 5755) is used for the knife-edge section of labyrinth seals, and stainless steel (AMS 5613) for the seal lands.

The high pressure fourth stage air and lubricating oil are obtained from transfer tubes routed through the diffuser case struts. The scavenge oil is returned by gravity through the diffuser case strut, located 45° from the bottom vertical centerline, to a scavenge pump in the gearbox. The number 2 and 3 bearing compartments are vented through tubes located in diffuser case struts. Thermal blanket insulation is attached to the outer walls of the compartment to reduce the temperature of the number 2 bearing compartment and to minimize heat rejection to the oil.

Two towershafts are required to transmit power from the engine. One shaft drives the side-mounted engine accessory gearbox. For the Boeing installation the second shaft terminates at a pad located at the top of the engine. For the Lockheed installation the shaft drives a right-angled, stepped-down gearbox and decoupler mounted at the same location. Starting is accomplished through the top towershaft for both installations. The spiral bevel pinions for the drives, located in the number 2 bearing compartment, are supported by ball and roller bearing assemblies bolted into the inner box structure.

The number 3 bearing compartment is supported by a conical section that transmits bearing loads to a conical case which mates with a flange attached to the inner diffuser case. The number 2 roller bearing is preloaded similar to the number 1 bearing. The bearing material is PWA 724 or PWA 725 and the conical case material is Waspaloy (PWA 1030).

The number 3 bearing compartment is oil-jet cooled and lubricated with a seal system similar to the number 2 seal. It consists of a carbon face seal that is loaded with fourth stage compressor discharge air and backed up by a labyrinth seal system. This limits leakage in event of carbon seal failure and protects the seal from exposure to high temperature, high pressure air. The seal schematic is shown in Figure 2B-58. Hastelloy W (AMS 5755) is used for the knife-edge section of the labyrinth seals, and stainless steel (AMS 5613) is used for the seal lands.

The number 3 bearing compartment is protected against high temperatures by thermal blanket insulation attached to the outer wall of the compartment.

Oil supply lines for the bearing compartment, venting lines for labyrinth seals, and high pressure air lines are routed through the diffuser case struts. Accumulated oil in the compartment is removed to the number 2 bearing compartment by an oil ejector type scavenge pump.

(2) Seals

The bearing compartments are sealed by a primary carbon face seal. Experience gained on the JT11D-20 has shown that a dry face type seal operates satisfactorily for high temperature seal applications. In a dry face seal, the oil flows directly through the seal plate for cooling. The primary advantage of the dry face seal is the reduction of oil leakage past the carbon seal.

The carbon seals are backed up by labyrinth seals. The chamber between the inner two sets of labyrinth seals contains fourth stage compressor discharge air to protect the carbon seal from high temperature air, to limit the oil leakage, to reduce the net seal loading, and to provide a longer life. The carbon seals are held against the oil-cooled seal plate by a spring. The seal plate is flame-plated with chrome carbide (PWA 50).

Bellows seals, in place of the spring and piston ring secondary seal design, are also being considered. The bellows seals would eliminate secondary piston ring seal leakage. This is important from the standpoint of oxidation and fire resistance. Figure 2B-59 presents the bearing compartment design now being evaluated.

The preceding compartment configurations are being designed for initial engine operation while a hydrostatic type seal will be developed for long life engines. This is a face seal which rides on a thin cushion of air a few ten-thousandths of an inch off the seal plate. The cushion of air is provided by an air supply from a pressure source greater than that to be encountered on either side of the seal.

The advantages of the hydrostatic seal over a carbon faced seal are:

- Reduced heat generation and lube side heat input. The lube side heat input could be reduced as much as 25%.
- Long life because of the air cushion effect.
- Fail-safe design, such that it will operate as a carbon face seal if the air supply should fail.

A hydrostatic seal similar to that shown on Figure 2B-60 has been successfully tested for 150 hours on a rig simulating high Mach number conditions. At the conclusion of the test this seal exhibited no measurable wear and the seal faces were absolutely clean. However, this seal does require a sophisticated pressure balance system which will require development.

b. Lubrication

The number 1, 2, and 3 bearings are lubricated by a combined oil mist and jet oil system and cooled by oil flowing through slots in the bearing inner race. In the number 2 and number 3 bearing compartments, the oil supplied to the inner races is picked up from stationary jets by rotating scoops. In the number 3 compartment, the oil flows through a series of slots to cool the inner race, and then flows radially out to the bearing compartment through a series of holes in the seal rub plate. The number 2 compartment oil also flows outward through a series of radial slots in the split inner race of the bearing to centrifuge the oil into the bearing for lubrication.

## 8. AFTERBURNER AND PRIMARY NOZZLE

### a. General Description

The afterburner for the STJ227 turbojet engine consists of a diffusing section, a combustion section, a fuel distribution and flameholder system, and a variable area exhaust nozzle. This assembly is bolted to the rear flange of the outer turbine exhaust case.

The diffusing section contains an inner cone and an outer diffuser case, and provides an efficient transition from an annulus to a single full-flowing cylinder by increasing the duct cross sectional area at a modest diffusion rate. The inner cone and heatshield bolt directly to the rear flange of the inner turbine exhaust case. The outer diffuser case is also slightly conical and includes the ejector mount ring. The case has installation provisions for the streamlined sprayrings and integral flameholders, the igniter, and the fuel drain plug.

The aft two structural cases of the afterburner form the combustion section. The cases are protected by an axially-segmented, corrugated, nonstructural liner with a heatshield cooled convectively by turbine exhaust gas flowing between the liner and the heatshield.

The use of a close-coupled fuel distribution and flameholder system is made possible by the high gas stream temperature, which provides good combustion efficiency in a short burning length and allows flame-holding from integral sprayrings and flameholders. Five combination sprayring/flameholder rings inclined aft are used. For high augmentation fuel is injected from four streamlined sprayrings located immediately upstream of the integral units. External fuel manifolds feed both zones. Ignition is provided by an electrically-ignited torch type igniter located upstream of the fuel system. Successful experience with catalytic ignition has been obtained in the JT11D-20 engine, and this will be considered as an alternative ignition source.

The variable area exhaust nozzle consists of a series of flap segments hinged from a supporting ring attached to the afterburner outer rear case. The flaps are operated by a bellcrank linkage system with a unison ring to ensure synchronization. The unison ring is actuated by eight hydraulic cylinders shown in Figure 2B-61. Cooling air from the liner discharge is directed at the nozzle flaps to keep them from overheating. A sketch of the afterburner system is shown in Figure 2B-62).

b. Physical Structure and Materials

One of the most important aspects of any afterburner is the fuel distribution and flameholder system. Experience on the JT11D-20 engine has shown that a reversed slope system has several advantages. By angling the sprayrings and integral flameholder rings aft into the expanding diffuser, a greater effective area results. This reduces the pressure drop across the system. Furthermore, the air is guided into the void area behind the diffuser cone and away from the liner. This system has demonstrated stable efficient combustion.

Fuel is introduced into the combustion chamber through two separately fed zones. Since this engine is being designed for long-life operation, fixed area orifices are being used at the present time. However a design study is in progress to investigate the possibility of using variable geometry sprayrings. The high overall turndown fuel flow ratio makes it necessary to utilize a staged fuel system with either type of orifice. For lightoff and partial thrust augmentation, fuel is injected from the close-coupled zone. Structurally, each of the five rings in this zone consists of a circumferential fuel tube enclosed by a V-gutter. Five equally spaced external trunks feed each of these rings individually. The streamlined heatshielded sprayrings which form Zone II are supplied fuel by five radial external trunks. The trunks which feed each zone act as support struts for the other zone.

To handle the high pressure (850 psi max) in the fuel system, Waspaloy (AMS 5706 and AMS 5544), a high strength-to-density ratio material, is being used. Pratt & Whitney Aircraft experience has shown that for the support structure L-605 (AMS 5537 and AMS 5759) is superior because of its stiffness and creep strength. Since this material resists buckling, it can better withstand the thermal stresses in the flameholder rings associated with gradients produced by the attached flame. The Zone II spraybar heatshielding is made of Hastelloy X (AMS 5536 and AMS 5754) which has been proven by test experience on the JT11D-20.

The four sections of the afterburner liner are the fuel baffle, the double-walled segmented front section, the double-walled segmented rear section, and the double-walled conical ramp section.

The fuel baffle forms the forward section of the liner and prevents fuel from entering the cooling flow annular area. A cross section of the liner front and rear segmented section shows a series of valleys and arches (Figure 2B-62). These valleys are beneficial for screech suppression. The forward section absorbs the periodic combustion energy fluctuations and prevents random pressure fluctuations from developing into cyclic

vibrations of large amplitudes. The ramp section, which directs cooling air at the nozzle flaps, is bracketed to the case at the front end and guided by bumpers at the rear.

The liner is to be constructed of TD nickel because of its ability to withstand temperatures at or above 2000°F for long periods of time. Hastelloy X (AMS 5536 and AMS 5754) is used for the less severely heated heatshield and the fuel baffle because of low cost and successful experience with this material in similar applications.

The diffuser cone is made from Hastelloy X (AMS 5536 and AMS 5754) and consists of a forged flange welded to a sheet metal cone with a stiffening ring at the rear end. The diffuser overall area ratio is 1.50 and the equivalent conical angle is 9.5°. These parameters are determined by the slope and length of the cone. The diffuser cone is shaped to account for the extra diffusion of the exit guide vanes and the blockage of the fuel distribution and flameholder system.

The afterburner case consists of the diffuser case and a two-section combustion chamber case. These cases contain the turbine exhaust gases and the afterburner combustion products. As they are the most highly stressed, the cases are made from Waspaloy (AMS 5706 and AMS 5544). Similar usage on the JT11D-20 has shown Waspaloy to be particularly suited to this application. The diffuser case is conical in shape and the two combustion chambers are cylindrical; each of the cases fastens to the adjacent sections at flanges.

The afterburner nozzle controls rotor speed by varying the nozzle area from 7.0 to 12.7 sq. ft. Nozzle areas are scheduled by the afterburner fuel control as a function of engine inlet temperature and pressure. The nozzle is octagonal in shape and consists of an assembly of eight flap type segments with eight interspaced seals. The flap segments and seals are cast of a corrosion-and heat-resistant nickel-base alloy (IN 100). Stiffening ribs are cast on the back side of both the flaps and the seals. The flaps and seals are attached to the afterburner rear combustion chamber case and may be individually removed for inspection or replacement (Figure 2B-61).

A "slave" link is attached to the rear of the main flaps on the same axis as the seals. The "slave" link is forged from a nickel-base alloy. Cam follower rollers mounted on the "slave" link fit into a mating channel track located on the outer edge of each flap. As the "slave" link is actuated, the rollers transfer the force into each of the two adjacent flaps. The geometry of the flaps and "slave" link is such that the cam follower rolls in the crack and maintains line contact at the mating surfaces.

The "slave" link and seal segments are pinned on the same axis. The seals are then free to rest on the front face of the flaps and to provide the sealing surface at the corner of the octagonally shaped nozzle.

The seal is positioned by the pin at the base and also by a tab-shot arrangement located on the "slave" link. This system allows the seal enough freedom to adjust to any alignment irregularities between adjacent flaps. The tab-slot system also keeps the seal from falling forward during nozzle operation when the engine is not operating. During normal engine operation the gas stream flowing through the nozzle forces the seal against the flaps. This "slave" link-flap system also synchronizes all eight flaps.

The "slave" link is connected to the bellcrank by a forged connecting rod. This link is forged from a nickel-based alloy with a threaded clevis on one end to provide the necessary adjustment during assembly.

The bellcrank is pivoted on a mount pad located on the inside of each of the eight struts. These eight struts are trapezoidal-shaped with a slot cut in the inboard face to allow clearance for the bellcrank and connecting rod. Since the bellcrank linkage is located on the inside of the strut, access holes are provided on the sides of each of the struts for assembly.

The bellcrank is actuated by a link connected to the unison ring. The unison ring is fabricated from sheet metal with stiffeners added with adjustable cam followers mounted on these stiffeners. The rollers are guided by tracks which are integral parts of the struts. The ring is positioned between the struts and the O.D. afterburner. Eight equally spaced arms extend radially from the unison ring into the center of each strut. Tierods extend rearward from the radial arms, connecting the unison ring with the bellcrank, and forward to the hydraulic actuator. Slots are cut in the struts to allow proper clearance. The advantage of this system is that normal actuator loads are not transmitted through the unison ring. The ring is loaded only in the event of an actuator failure. Any unbalanced loads are then transmitted through the unison ring.

Eight actuators utilizing JT11D-20 design experience are mounted on the forward face of the mount ring. Under normal cruise conditions all linkages of the nozzle except the "slave"link connecting rod are loaded in tension. This eliminates the buckling problem normally found in the tierods.

Engine experience gained on the JT11D-20 has shown that the IN 100 material chosen for the flaps can withstand the temperature environment in the STJ227 afterburner.

c. Temperatures and Stress Levels

The temperatures of the afterburner structural components are regulated to obtain a minimum weight design. Structural temperature control is achieved by utilizing turbine exhaust air for cooling and by metal heat-shielding. Available materials determine temperature limits.

The optimum combined weight of the cases and the liner cooling system occurs with a maximum case temperature of 1400°F. Additional cooling results in slight increases in case strength with undesirable increases in heatshielding weight.

The segmented liner configuration chosen (Figure 2B-62) provides a liner and cases which are subjected only to low tensile loads, resulting in a lightweight design. Using the cooling air available (5% of turbine exhaust gas) and keeping the case temperature at a maximum of 1400°F, the resulting temperature of the liner is 2000°F and that of the heat-shield is 1600°F. Further cooling of the liner provides no weight advantage, as liner thickness is limited by manufacturing. Increasing the amount of turbine exhaust air used for cooling results in a performance loss associated with increasing the amount of unburned air. The area between the heatshield and the combustion chamber case is vented to the turbine exit pressure but has no flow. This cooling method results in temperatures that permit the use of existing high temperature materials.

Operating temperatures of the nozzle flaps are controlled by convectively cooling the flaps with the liner cooling flow. Allowable flap temperatures in the range of 1800°F were considered in choosing the amount of cooling air flow for the system.

Maximum loading of afterburner components occurs at sea level take-off when maximum flow and maximum internal pressures occur. For this reason, most afterburner components are limited at sea level take-off and must have a design life at these conditions equal to the expected sea level take-off operating time in 10,000 hours engine life.

d. Afterburner Flap Loads

The nozzle area is controlled by an actuator system that is tied into the fuel control. The forces necessary to control the flap areas are determined primarily by the main gas stream loading on the flaps.

To calculate these loads the flap is first segmented into a series of increments, and a static pressure profile is established using isentropic relationships as a function of the total gas stream pressure and the cross

section area ratio between the increment location and the throat. The incremental load ($F_N$) is found by using a ring area based upon the incremental width and the radius to the increment. The flap load calculation is shown in Figure 2B-63. Moments are then summated about the center of the flap hinge (0) and set to equilibrium: $\sum M_O = \sum F_N D_N = F_R D_R = 0$. This establishes the reaction load ($F_R$) at the point of contact with the flap actuator roller. This roller load is then used to calculate the required pressure levels in the flap actuator system.

A computer program has been written that takes the increments of the flap and integrates the pressure profile and its corresponding load. Using this integrated pressure load exerted by the gas stream, the program then solves for the resultant force that determines the actuator force system.

All of these procedures have been established neglecting friction in the system. The values of friction can be determined only by testing. The flap is held stationary by a combination of actuator and friction forces, which are always in the direction opposing motion. To move the flap in a direction opposite to the natural movement, the flap actuator must overcome this friction force. After an equilibrium point is established the actuator stabilizes at a new load level. By running a transient test, observing the load level of the actuator system, and comparing to steady-state load levels, the friction forces can be determined. These friction forces can be incorporated into the flap loads and their corresponding actuator load requirements.

The JT11D-20 afterburner flap systems have been tested for both transient conditions (i.e., opening and closing the nozzle flaps) and steady-state conditions. Data obtained substantiated both gas stream loading calculation procedures and friction force estimates at these conditions. These data have been obtained by inserting pressure taps into the actuator system, measuring pressure levels necessary for actual operation, instrumenting the actuator arms with strain-gages, and measuring the deflection loads being transmitted to the flaps. Design conditions and operating stress levels are summarized below.

| Afterburner Component | Operating Temp °F | Critical Operating Condition | Material | Weight Limiting Factor |
|---|---|---|---|---|
| Cone | 1550 | SLTO | Hastelloy X | Manufacturing |
| Flameholders | 1550 | SLTO | L-605 | 0.5% creep at SLTO life |
| Sprayrings | 600 | SLTO | Waspaloy | 0.2% yield int. pressure |
| Ducts | 1400 | SLTO | Waspaloy | S.R. SLTO life |
| Heatshield | 1600 | SLTO | Hastelloy X | Manufacturing |
| Liner | 2000 | SLTO | TD Nickel | Manufacturing |
| Flaps | 1800 | SLTO | IN 100 | 0.5% creep in SLTO life |

9. OCTAGONAL BLOW-IN DOOR EJECTOR-REVERSER

a. General Description

Design studies of various types of ejectors during Phase I and Phase II have resulted in a new lightweight, simplified ejector configuration. This configuration, based on extensive wind tunnel testing and combining the best mechanical features of previous ejector designs, meets the stringent aerodynamic requirements of a subsonic and supersonic ejector while incorporating an efficient and flexible reverser system. A schematic drawing of the ejector-reverser system is presented in Figure 2B-64.

b. Configuration

(1) General

The ejector, as shown in Figure 2B-65, is a regular polygon, rounded at the corners. For this study we chose eight sides. However, if future studies are required, the ejector could have as many as sixteen sides and still retain the inherent design features. The ejector consists of two main parts, the fixed structure and the translating shroud. The fixed structure contains the blow-in doors, actuation system, and reverser cascades. The translating shroud contains the reverser doors, outer skin, and trailing edge flaps.

(2) Support Structure

Eight crossbraced struts form a stationary framework to support the ejector-reverser assembly. Each strut is trapezoidal in cross section and occupies 10° of circumference. Lips on the outer edges of the struts provide support and sealing for the blow-in doors (Refer to Figure 2B-65). Additional intermediate seals between adjacent doors provide a "double seal" configuration at cruise conditions. These lips and seals add 1° of additional blockage to tertiary airflow area, making a total circumferential blockage of 88°. This is a reduction of some 20° from models tested during Phase I.

(3) Blow-In Doors

Eight free-floating double-hinged blow-in doors are hinged to the support structure at their leading edge and are supported by the struts. "In" stops are provided to limit door travel and to provide the correct contour for blow-in flow. At the maximum "in" position, the doors "tuck" behind the afterburner nozzle hinge plane, providing a smooth flow passage into the ejector throat.

(4) Actuation System

The actuation system for the shroud translation and reverser door operation consists of four 2-position hydraulic actuators mounted internally in alternate struts. This feature of the STJ227 is shown in Figure 2B-66. The rod end of each actuator is connected to the translating shroud support structure.

(5) Reverser Cascades

Reverser cascades are mounted between the support struts immediately behind the trailing edge of the blow-in doors. Each airframe installation, as shown in Figure 2B-67, has different reverse targeting patterns. These patterns are preliminary and can be easily revised to suit final airframe requirements. The space between struts is blocked where reversal is prohibited. As shown in the Boeing configuration (Figure 2B-67, reverse flow can be directed at an angle, or normal to the flat sides of the ejector as desired.

(6) Translating Shroud

The translating shroud forms the inner contour of the ejector. As shown in Figure 2B-66, the shroud consists of eight struts which connect the front ring to the rear support ring forming a movable shroud framework. The reverser doors are pivoted on the sides of the struts, while the outer skin and trailing edge flaps are mounted on the rear support ring. The entire movable structure is mounted on roller supports, as shown in Figure 2B-65, and guided in tracks on the fixed struts.

(7) Reverser Doors

As shown in Figure 2B-67), four large reverser doors comprise approximately 75% of the blocked area and four small triangular doors make up the remaining 25%. This configuration has three inherent advantages. Structurally, the major reverser loads are transmitted to the octagonal structure normal to each alternate side at the struts, eliminating bending loads in the end frame. Aerodynamically, this configuration allows null-thrust operation by blocking a portion of the flow with the large doors, while the small doors can remain open allowing normal flow to counterbalance the blocked reverse flow. Mechanically, this configuration allows the large doors to be rotated into position before the small doors complete the flow blockage, thereby reducing actuator forces.

(8) Outer Skin

The outer skin "seals" the ejector at cruise and provides a smooth cover for the reverser cascades. The skin is connected to the rear support ring of the movable shroud and is supported from the fixed struts by four rollers at each of the eight corners.

c. Operation

(1) Forward Flight

For all forward flight conditions, the ejector is actuated aerodynamically. During the subsonic portion of the flight profile, the blow-in doors are "in" and the trailing edge flaps "close" providing a convergent nozzle configuration. At supersonic conditions, the blow-in doors close and the trailing edge flaps open to form a divergent nozzle configuration.

(2) Subsonic Reversing

For reversing during "on-the-ground" or subsonic operation, the actuators translate the shroud aft, exposing the reverser cascades. As the translation progresses the reverser door links, which were previously held in a locked position, "bottom out" at the rear of the fixed structure. As the shroud continues translating aft, the door links start rotating the large reverser doors into the engine flow stream. At the last portion of the translation, door links on the small triangular doors also "bottom-out," rotating the small doors into the engine flow stream. This completes blockage and establishes full reverse flow.

At the start of the subsonic reverse cycle, the blow-in doors are "in". As the large reverser doors enter the air stream, the ejector "pumping action" is spoiled and secondary flow pressure from the inlet closes the blow-in doors. Flapper valves in the secondary air passage immediately upstream of the blow-in doors close as soon as ejector pressure rises above inlet pressure prior to reversing. This prevents hot reverser gases from flowing forward into the control compartment or engine inlet. On the Lockheed installation, these valves are mounted between the nacelle and the afterburner duct. On the Boeing installation, they are mounted integrally in the secondary air bypass ducts. In both cases the valves are free-floating and close only during reverse pressure conditions.

(3) Supersonic Reversing

For reversing during supersonic flight conditions, the same sequence takes place except the blow-in doors are already closed. Therefore, only the secondary flow flapper valves would close on complete reversal.

(4) Null-Thrust

A null-thrust condition can be achieved by replacing the 2-position actuator with a 3-position or variable position actuator. Since the large reverser doors can be rotated into position before the small doors close, a scheduled position for the shroud could achieve null-thrust. As before, the portion of the flow reversed would automatically close the blow-in doors and flapper valves. The translating shroud would move aft only a portion of its full travel. In this same configuration with a variable actuator, a full range of reverse thrust could be achieved without changing the engine power setting. This would require an additional lever in the cockpit such as that represented by the cut-off lever with modulation capability as proposed in Phase I.

(5) Fail-Safe Feature

At all times, the reverser system is fully synchronized due to the "synchronizing ring" effect of the shroud framework. The reverser system is designed "fail-safe" by locating the door links slightly below the center of pressure. If complete hydraulic failure occurs, the reverser doors fold down out of the main engine stream. As the ejector regains its "pumping action" the trailing edge flaps close, and forward thrust is exerted by the differential pressure across the flaps, driving the shroud forward into the forward flight condition. At supersonic conditions, if failure occurs during reversal, the shroud would move a portion of the translation forward, and the ejector performance would be reduced until subsonic flight conditions are reached. At that point the shroud would move full forward and the ejector would perform as in normal forward flight.

d. Physical Structure and Materials

(1) Fixed Support Structure

The eight fixed struts are hollow, trapezoidal in shape, and fabricated from sheet Waspaloy (AMS 5544). The inner face has a forged Waspaloy track the full length of the shroud framework. The tracks are machined after complete assembly to assure alignment with the

movable shroud structure rollers. The eight struts are welded as a unit to a rear Waspaloy channel, which forms an octagonal frame. Cross-bracing rods are located between each of the struts, forming a braced structure from the rear channel frame forward to a point just aft of the trailing edge of the blow-in doors. These rods are pre-loaded and welded to the struts. The struts then cantilever forward from this structure and terminate at ball joint connections on the afterburner duct ejector mount ring. This cantilever feature permits thermal incompatibility between the ejector and the afterburner duct. The ball joint connections eliminate bending loads at the mount plane. This allows the struts to become simple tension-compression members.

Each strut as shown in Figure 2B-69 has an intermediate seal on its outer face in the area of the blow-in doors. Besides providing an additional seal for the blow-in doors, this seal assures that the entire strut is sealed within the secondary flow path, eliminating thermal gradients across the strut.

On the rear of the fixed structure, at each side of each strut, tracks are provided for holding the reverser door links in a locked position. A stop is also provided at the extreme end to lock the links in position for reversing.

Provisions for internally mounting the reverser actuators in alternate struts are incorporated. The forward portion of each strut has provisions for the actuation system for the variable afterburner nozzle.

(2) Blow-In Doors

The eight blow-in doors are flat panel double sheet honeycomb construction. The forward half of each door has forged hinges to connect with the struts. The rear half of each blow-in door is hinged to the front half and has trailing edge stiffener sections. Each door is designed as a flat plate under uniform pressure, simply supported along two sides. The highest pressure load across the doors occurs during "on-the-ground" reversal when temperatures are not severe. At cruise conditions the differential pressure is 20% of maximum reversal pressure. This allows the doors to be designed yield-limited at low temperatures. Titanium honeycomb construction is used for all blow-in doors.

(3) Reverser Cascades

The reverser cascades are mounted between the fixed struts immediately aft of the trailing edge of the blow-in doors. In areas where reversing

is prohibited, honeycomb panel sections replace the cascades. The leading edge of each cascade forms the outer portion of the blow-in flow ramp to the ejector throat. The trailing edge of each cascade forms a smooth ramp with the movable shroud front ramp when translated to reverse position. Between the front and rear faces a series of air guide vanes direct reverse flow to precise targeting requirements. Each cascade or panel section is a complete structure, and bolts to the sides of the fixed struts. Construction will be fabricated sheet metal Waspaloy (AMS 5544) for the cascades, and Waspaloy honeycomb paneling for the blocked portions.

(4) Actuators

Four hydraulic 2-position actuators are mounted internally in alternate fixed struts with the rod end connected to the movable shroud rear support ring. These struts provide a cooling air passage and heatshield for the actuators. The actuators are fabricated primarily of 17-4 PH stainless steel (AMS 5643) to provide the proper strength and wear characteristics. Fuel, cooled by a continuous recirculation system, is supplied to the cylinders from the reverser control.

Sealing is accomplished by a 2 section shaft seal with a drain between the sections. The seal housing is designed such that recirculating fuel protects the seals from high temperatures. The seals are protected from contaminants by the cast iron bushing in which the shaft rides. The bushing has two grooves to trap contaminants. The actuator cylinders and seal housings are designed to withstand fluid pressure surges.

(5) Translating Shroud

The translating shroud consists of a movable shroud framework and a liner. The framework consists of eight struts connecting the front support ring with the rear support ring. Between each strut, in the areas not occupied by reverser doors, interconnecting ribs provide support for the liner. All struts and the front ring are fabricated Waspaloy hollow beam sections. The rear support ring is formed and fabricated from Waspaloy sheet. Forged Waspaloy bosses, as shown in Figure 2B-65, are welded on the sides of each strut to provide pivot points for the reverser door hinges, and on the outer faces to provide mountings for the support rollers. The entire structure then forms a rigid framework for the reverser doors, liner, outer skin, and trailing edge flaps. Rollers on each strut provide support for the movable structure. Each roller is of caged roller bearing construction,

specifically designed to withstand intermittent shock, high static loads, and high temperatures. The roller is designed to operate without lubrication, and the races and rollers are fabricated of Waspaloy. This type of roller has been used successfully on the afterburner flaps and synchronization ring for the JT11D-20 engine. All rollers are mounted radially to maintain alignment of the ejector while permitting thermal incompatibility during transients.

The liner forming the inner contour of the ejector throat consists of 16 sections. Each flat side of the octagon is a separate sheet riveted to the shroud support structure to allow for thermal expansion. Each corner of the octagon is a separate seal section, supported from the movable struts, sealing either to the liner or the reverser doors. This construction allows thermal growth of the throat without radial growth. The correct aerodynamic contour over the full exhaust temperature range is therefore maintained. The liner shields the rigid support structure from the hot exhaust gases and reduces thermal problems on the structure. The liner sections are made of Hastelloy X (AMS 5536), and are designed for easy replacement at scheduled overhaul periods, lowering the time and cost of overhaul.

(6) Reverser Doors

All eight reverser doors (both shapes) will have a rigid structural framework supporting a lightweight liner similar to the translating shroud. As shown in Figure 2B-66, the support structure bridges between the struts, and the liner seats against the corner seals on the movable shroud. At the door structure midpoint, a pair of door links is mounted to the sides of the structure. These links are free to pivot in the structure with the trailing end of each link guided in a track on the sides of each fixed strut. A locking rail is provided on the movable shroud struts to hold the link in the locked position during the first part of shroud translation. The door and link move as a unit until the link trailing end strikes a stop at the rear of the fixed structure. A notch in the fixed strut track allows the link to drop down and the movable strut locking rail passes over the link end. This holds the link in position for pivoting the reverser doors into position. The link is in tension while the doors are in stowed position. This removes the load from the track on the fixed strut, and eliminates a dragging condition on the link end during translation. When the link bottoms into the fixed structure, it rotates in a socket as the reverser doors open. The elimination of drag on the link allows the use of a solid end without rollers, bearings, or bushings. The link then becomes a compression member as doors rotate into position and reach a maximum load condition when doors are full open. Approximately 90% of the reverser

load is transmitted directly to the fixed structure rear framework. As stated earlier, due to the door configuration, this load is applied normal to alternate sides of the rear octagonal ring at the strut locations. This eliminates bending of the ring due to punch loads. The remaining reverser load is transmitted to the door hinge point providing a forward thrust component on the movable shroud to achieve the "fail-safe" feature. All links will be forged Waspaloy (PWA 1007). The door structure will be forged Waspaloy I-beam welded into an integral grid construction. This structure is riveted to and supports the door liner. As in the translating shroud, all door liners are designed for easy replacement at scheduled overhauls. Liner construction will be formed Hastelloy X sheet (AMS 5536).

(7) Outer Skin

The titanium outer skin consists of a smooth outer sheet reinforced by a corrugated inner sheet. As shown in Figure 2B-65, the skin shape is octagonal with rounded corners. The leading edge is a forged ring, and the trailing edge is flanged for bolting to the movable shroud rear support ring. At four locations along its length, at each corner, the outer skin has rollers to provide support and stand-off from the fixed structure. Each roller is bolted to the skin and is similar in construction to the translating shroud support rollers.

(8) Trailing Edge Flaps

The trailing edge flaps are of two basic shapes. The corner flaps cover the full radius and extend into the flat area sufficiently to overlap the flat flaps. The flat flaps dovetail into the corner flaps. This construction allows flat sliding surfaces between adjacent flaps. The hinge ends of the flaps, as shown in Figure 2B-66, have an end channel with forged hinge pads. Each flap has two clevis joint type hinges, one at each side of every flap for maximum wheel base. Seal plates bolted to the end channels provide spring seals for the hinge line. The inner surface of each flap is Hastelloy X (AMS 5536) sheet construction similar to the shroud liner; the outer surface of each flap is Waspaloy (AMS 5544) sheet similar to the inner surface. Connecting these two surfaces is a series of corrugated Waspaloy shear webs running the length of each flap and riveted alternately at corrugations to the inner and outer sheets. This construction provides a rigid shape for the flaps, but allows the inner sheet to expand relative to the outer sheet. The inner and outer sheets are welded to the hinge channel section but they are separate at the trailing edge. The resulting assembly consists of sixteen large flaps with no loose seal plates. Seal surfaces are flat and do not have to adjust to changing contours. The hinge loca-

tion at the inside of each flap allows pressurizing the space between the shroud end support and flap hinge channel with secondary air pressure. The high secondary air pressure tends to close the flaps. This feature can be utilized to prevent overexpansion of the exhaust gas stream at certain flight regions.

c. Temperature and Stress Levels

The ejector is designed to utilize either secondary air or blow-in door air for cooling throughout the main structure. Maximum temperatures will be at supersonic cruise conditions. With a secondary air temperature of 575°F, the fixed structure blow-in doors, and shroud structure area should not exceed 700°F. During reversing, however, a portion of the fixed struts and reverser cascades will be subjected to 1200°F airflow. The structural analysis is based on this temperature condition and stress-limited to design allowable yield. The liner surface of the shroud and trailing edge flaps reaches a maximum temperature of approximately 1650°F during supersonic acceleration. During cruise conditions this maximum condition should decrease to approximately 900°F. Stress analyses were based on the maximum temperature condition. The liner, being a series of simply supported small panels, is yield-limited at high temperature, and creep-limited at cruise. The reverser door liners receive a substantially higher pressure during reversing, and are yield-limited at the reversing temperature of 1200°F. The blow-in doors are yield-limited at the reversing pressure. However, since no flow occurs over the doors during reversing, the temperature is reduced to 800°F. The following table summarizes materials and design limitations of the components.

MATERIAL AND DESIGN LIMITATIONS

| Component | Material | Limited |
|---|---|---|
| Translating Structures | Waspaloy | Creep |
| Inner Skin | Hastelloy X | Creep |
| Outer Skin | Titanium | Yield |
| Floating Flaps | Waspaloy and Hastelloy | Creep |
| Cascades | Waspaloy | Yield |
| Blow-In Doors | Titanium | Yield |
| Reverser Doors | Waspaloy and Hastelloy X | Yield or Stress Rupture |
| Support Structure | Waspaloy | Buckling |

All material, where stress-limited, will be in accordance with design allowable stresses as specified in the Phase II-A report.

## 10. CANTED ENGINE CONFIGURATION

### a. Lockheed

Both airframe manufacturers have requested that the engine be bent in such a way that the aft section is canted downward. Lockheed has requested approximately a 4° bend in order to give a downward component of thrust.

The bend was placed in the area just aft of the turbine exit guide vanes, and upstream of the afterburner fuel system sprayrings. Bending in this area gives a long straight section between the bend and the nozzle. This allows the flow enough distance to make the turn to the desired angle. Bending upstream of the afterburner duct and nozzle area allows all eight nozzle flaps, the nozzle flap actuators, and the bellcrank systems to be identical.

Certain problems exist in the bent engine that would not exist if it were straight. The hot gas flow must be turned by the liner on the upper side of the engine. This results in increased temperature in the liner at the top of the engine and a colder section at the bottom. The hot section can be sufficiently cooled by using supplemental cooling. This is accomplished by additional cooling air quantity or velocity to increase the convective cooling locally. Film cooling in the hot section could be employed as an alternative.

With the bend located in the turbine exit guide vane area, it is in an optimum portion of the duct between the rear engine mount and the ejector mount. The struts are straight since they are aft of the engine bend plane.

The blow-in doors are built into the ejector and are not part of the airframe. All eight doors are identical and are open to admit tertiary air to the ejector during subsonic flight. During supersonic flight and subsonic reversing, the doors are closed. The secondary and tertiary air ramp to the ejector is inclined at the same angle to the ejector centerline on all eight sides. The cant of the engine causes a drag force on the bottom blow-in doors and a reduced outside pressure on the top doors. This may result in irregular closing of the doors during transonic acceleration. However, wind tunnel testing has shown that unequal blow-in door positions have little effect on ejector performance. This effect is stablized at conditions other than transonic acceleration.

The reverser operates by forcing air through the ejector cascades in a forward direction. Any effect on reverser performance resulting from canting the engine can be corrected by proper selection of reverser air exit locations. Cascade areas not needed for reversing are blocked.

b. Boeing

Boeing has not defined the amount of bend in the engine, but has established ground-rules concerning the position of the engine with respect to the wing. These govern the position and amount of bend in the engine. The position of the centerline of the engine inlet is fixed. The engine must extend at least 3 inches, but not more than 6 inches, above the wing at the trailing edge. The ejector centerline must pass through a specified target in the stabilizer region of the airframe. The bend in the engine is determined to be 8° 30' based on present requirements. The fore and aft bend in the engine also depends upon the airframe requirements. The bend must lie in the area of the nozzle, and it was determined that it must be at the maximum area position of the nozzle for this installation.

Because engine performance is greatly affected by nozzle performance, a two-dimensional exhaust nozzle angled downward is being investigated on an open water table using the hydraulic analogy between a compressible fluid and an open channel water flow. The preliminary hydraulic analogy investigation is to define problem areas. A more extensive investigation is required to solve the problems uncovered and to establish basic design criteria for this type of nozzle design.

A number of configurations angled from 8° to 16° have been tested for the afterburning turbojet nozzle. Preliminary results indicate that adequate turning is achieved at the low duct Mach numbers occurring during minimum nozzle area operation. A problem exists in achieving full turning at the higher duct Mach numbers encountered at maximum nozzle area operation.

Early testing has indicated that the duct wall design influences flow; and, if some turning can be achieved in the duct, the rest may be accomplished at the nozzle. Figure 2B-68, I and II, shows initial duct wall designs tried. Figure 2B-68, III, shows an improved duct wall design with a 40% increase over Figure II in the amount of turning accomplished at the minimum nozzle condition. Figure 2B-69 shows four of the many water table photographs used in compiling data.

An area of increased temperature may also exist in the upper section of the Boeing liner. Supplemental cooling, described for the Lockheed engine, will be provided.

Canting the engine in the area of the nozzle for the Boeing installation requires that the structural struts be bent. Although a bend here will require additional structure within the struts for strength, it is not a major problem. Each of the eight struts will require a slightly different degree and direction of bend. All nozzle flaps are identical and are actuated by eight bellcrank systems. Each bellcrank system must be slightly different in movement and mechanical advantage to give uniform flap movement. The synchronizing ring transfers the equal actuator loads to the unequal flap loading requirements.

Seven of the blow-in doors are built into the ejector. The top door is attached to the airframe to form a continuation of the upper wing surface. All eight doors are different. The leading edge of each is perpendicular to the engine centerline, and the trailing edge mates with the leading edge of the secondary and tertiary air ramp. This ramp is perpendicular to the ejector center line and is therefore inclined relative to the engine centerline and blow-in doors. Blow-in door and reverser operations are effected in the same manner described for the Lockheed engine.

## 11. EXTERNAL PLUMBING, BRACKETS AND HARDWARE

### a. General Description

The plumbing and associated brackets-hardware designs utilize the latest methods used on the JT11D-20 engine to provide durability, accessibility, and ease of assembly. The materials used for plumbing, brackets, and seals are based on experience gained from the design and operation of the JT11D-20, since the high environmental temperatures associated with the STJ227 engine are quite similar.

The tubing materials are type 347 stainless steel (PWA 770) and Inconel (PWA 1060). Stainless steel is used for tubing that does not exceed 800°F when exposed to its individual functional and environmental temperatures. Clamp standoffs on these tubes are attached in accordance with PWA Specification 2666, Silver Brazing. Inconel is used for tubing that exceeds 800°F when exposed to its individual functional and environmental temperatures. Clamp standoffs on these tubes are attached by gold-nickel brazing (PWA Specification 19).

The plumbing lines are formed to minimize assembly misalignment tolerances. The lines are routed to avoid fluid traps, and provide calculated flexibility to accommodate misalignment tolerances and thermal expansions. In dry lines, where flexibility by line routing

is not practical, bellows or slip joints are used to allow for misalignment and thermal growth. The use of low emissivity coatings (gold coating, Specification PWA 55) to reduce heat absorption eliminates the need for mechanical heat shield. Mechanical joints are utilized to further simplify assembly and disassembly.

The plumbing system uses modified AN standard threaded connectors with fittings machined as integral parts of the tubes. This design has higher fatigue strength and greater reliability than tubes with brazed fittings, since the uncertainty of producing sound brazed joints is removed.

Plumbing clamps and brackets are provided to support the lines and to avoid resonant vibrations. The brackets are of fully heat-treated Inconel X (AMS 5542) material, spaced at calculated intervals. (The sliding and fixed bracket assemblies are presented in Figures 2B-70 and 2B-71 respectively.) Where necessary, brackets are formed and slotted to allow free movement of the line in the required plane to relieve thermal stresses. Wear-resistant materials and coatings are used on all sliding brackets to provide increased durability. In addition, vibration dampers are used on all high pressure lines sliding brackets, and where the calculated tube movements coincide with the plane of the maximum vibratory movements. The vibration damper, designed for the JT11D-20 engine, allows thermal growth, but restricts vibratory movement. The calculations for the tube thermal growths, routes, flexibility, supports, and clearances were obtained through an IBM computer program that was established and utilized in the design of the JT11D-20 engine plumbing.

1. System Description, Material Selection, and Routing

(1) Hydraulic System

The hydraulic plumbing system for the STJ227 engine is composed of the hydraulic supply and return lines for all hydraulic actuators. The hydraulic fluid used is engine fuel. A schematic of the system is given on Figure 2B-72.

The hydraulic pump is mounted on the rear of the engine accessory drive gearbox and supplies fuel to the system at 1500 psi. This pump provides fuel directly to the discharge filter, which is mounted aft of the pump on the right side of the engine. Fuel from the discharge filter is routed to the pilot valve in the main fuel control for inlet guide vane and bleed door actuating. From the pilot valve, tubes are routed forward to the starting bleed door actuator manifolds and

to the inlet guide vane actuator manifolds. Discharge fuel then passes to the exhaust nozzle control and reverser pilot valves. Tubes from these valves are routed aft to the appropriate actuator manifolds.

A portion of the discharge fuel is bled through an orifice and routed to the recirculation lines from the windmill bypass low idle, shut-off and drain valve and the combined fuel shut-off and recirculation valve. Fuel is supplied to these lines at the fitting on the valve to provide coolant flow to the system for non-afterburning operation. Filtered "make-up fuel" for the hydraulic system is provided by the main fuel pump. Make-up fuel is routed from the main fuel pump directly to the hydraulic pump inlet. This make-up fuel maintains the design fluid temperature in the system.

Each set of actuators is provided with an open manifold, a close manifold, a coolant return manifold, and a seal drain manifold. Tubes from each coolant return manifold are routed to a central return tube which leads to a return filter located near the discharge filter. A tube from the return filter tees into the make-up tube from the main fuel pump. The control valves also have return lines which connect to the central return tube.

The hydraulic pump, filters, and control valves are provided with threaded bosses and adapters as shown in Figure 2B-73. A metal gasket is used between the boss and adapter.

Stainless steel (PWA 770) is used for all supply and return tubes; the drain tubes are Inconcel (PWA 1060). Tube sizes were selected to provide sufficient flow and reaction time, with adequate wall thickness for the fluid pressure. All of the hydraulic system tubes utilize integral fitting mechanical connections as shown in Figure 2B-74. The tubes also incorporate expansion loops, where required, for thermal growth. A nickel conical gasket is used as a seal at the connections. The supply and return tubes are gold-coated to reduce heat transfer.

An IBM program developed for the JT11D-20 engine is used to determine the static stress in the tubes due to thermal deflections. This program is used for all plumbing systems, but it is of particular importance in the hydraulic system. In this system, low static stresses must be maintained to offset the stresses due to high fluid pressures and increased vibratory stresses. In the hydraulic system, the tubes are subjected to hydraulic excitation in addition to the engine-induced vibration. The IBM program provides a quick and accurate means to route the tubes for the best compromise of short lightweight lengths, while maintaining sufficient flexibility to keep the stress level low.

(2) Fuel System

The fuel system for the STJ227 engine is composed of the main burner fuel supply and the afterburner fuel supply. The main and afterburner fuel systems are shown in Figures 2B-75 and 2B-76 respectively.

A bifurcated supply tube to the main and afterburner pumps is provided, with the airframe connection located at the top of the engine, as shown on the installation drawings. The pumps and airframe connections use angle gasket type seals, with bolted flanges brazed to the tubes (Figure 2B-77).

The main fuel pump delivers fuel directly to the main fuel control; these units are located on the right side of the engine. Fuel is routed from the main fuel control to the main fuel-oil cooler located on the lower right side of the engine. The fuel is then routed to the windmill bypass low idle shut-off and drain valve located at the bottom of the engine. The fuel from this valve is routed through four manifolds to eight clusters of nozzles in the main burner system. There are four nozzles in each cluster, connected in series by three short jumper tubes. One nozzle in each of two adjacent clusters is used for the low idle system. Fuel is routed from the low idle valve to the low idle nozzles.

All the main fuel supply lines from the main fuel control to the windmill bypass low idle shut-off and drain valve employ bolted flange connectors brazed to the tubes with silver braze (AMS 2666). Angle gasket seals, shown in Figure 2B-77, are used at these connections. The main burner and low idle manifolds, as well as the nozzle jumpers, are integral ferrule tubes with modified AN connectors at the nozzles and valve. Conical nickel gaskets provide sealing at these joints. The integral ferrule tube and conical gasket are seen in Figure 2B-74.

The entire main fuel system is fabricated from stainless steel (PWA 770) tubing and is gold-coated for heat rejection.

In the afterburner system, fuel is routed from the turbopump to the afterburner fuel control. Bolted flange connections and angle gasket seals are used at both components. The tube material is stainless steel (PWA 770).

The Zone I fuel is routed from the control to the afterburner fuel-oil cooler, located on the lower left side of the engine, and then to the Zone I sprayring manifolds. The Zone II supply lines, however, are routed directly from the afterburner control to the Zone II sprayring

manifolds. All plumbing from the afterburner fuel control is fabricated from Inconel (PWA Specification 1060) tubing, with gold-nickel brazed (PWA Specification 19) bolted flange fittings.

At the forward end of the routing moderate thermal shock is encountered, therefore, angle gasket seals (Figure 2B-77) are used. At the aft end of the system and at the sprayring fittings, which are subject to extreme thermal shock, spring gasket seals are employed. A schematic of this sealing design is shown in Figure 2B-78.

The sprayring manifolds from the supply lines to the sprayring fittings contain "S" bends to compensate for high thermal gradients. In both Zone I and Zone II manifolds, tees are provided to permit draining of residual fuel through lines to two drain valves located at the bottom of the engine. These drain lines are of Inconel (PWA 1060) material, with gold-nickel brazed, bolted flange fittings that incorporate spring gasket seals.

Signal lines to the drain valves are routed from the afterburner fuel control. These lines are made of stainless steel (PWA 770) tubing with integral ferrules. Gold coating is used on these lines to reduce heat transfer. Conical nickel gasket seals are used at the control and valve connections.

Recirculation lines are provided for both main and afterburner systems, and are routed from the two supply lines to an airframe connection point on the engine. These lines, made of stainless steel (PWA 770) tubing, incorporate integral ferrules and are gold-coated. Conical nickel gaskets are used as seals.

(3) Lubrication System

The STJ227 engine lubrication system (Figure 2B-79) consists of oil pressure supply, scavenge, and breather lines. The engine lubricating oil is routed from the oil tank to the oil pump inlet port. The pressure oil from the pump outlet is routed through two fuel-oil coolers placed in series. Downstream of the fuel-oil coolers, a tee is provided in the lube line to accommodate a cooler bypass line from the oil pump bypass port. This tube also functions as a pressure sense line. The lube oil is then routed to a fitting which contains provisions for a temperature sensor and an oil pressure transmitter. This fitting is located on the right side of the engine. Immediately after the oil pressure transmitter fitting, the oil supply is routed to the number 1, 2 and 3 bearing compartments, and to the gearboxes.

The number 1 bearing oil supply line is routed across the compressor case to a strut on the compressor inlet case. The number 2 and 3 bearing oil supply is routed to a strut on the diffuser case. The main gearbox lube oil supply line is routed forward and over the engine to the main gearbox.

The main scavenge pump is located in the lower portion of the main gearbox, which is used as a sump. Scavenged oil is pumped directly from the gearbox sump to the oil tank. Scavenged oil from the number 2 and 3 bearing compartments passes through a diffuser case strut to the main gearbox scavenge manifold. The number 1 bearing compartment scavenge oil is routed from a strut on the inlet case of the compressor, across the compressor case, to the main gearbox scavenge manifold. The main gearbox scavenge manifold is then routed to the oil tank.

The lube pressure and scavenge lines are integral ferrule tubing, made from stainless steel (PWA 770) and gold-coated per PWA Specification 55 to resist heat transfer.

The engine breather system provides the oil tank and bearing compartments a regulated pressure of 8 to 15 psia at all flight attitudes. A breather manifold from the gearbox to the oil tank incorporates two tubes that are joined near the gearbox. One tube assures breather pressure at normal flight conditions; the other tube is for inverted flight. Tees are provided in the inverted flight tube for breather tubes from the number 1 bearing compartment and from the angle accessory gearbox required for the Lockheed installati . The number 1 bearing compartment breather connection is located at a strut on the compressor inlet case.

The breather tubes have integral ferrules, and are made of Inconel (PWA 1060).

Tube end-point connections, nipples, and tees employ the modified AN type connector, as shown in Figure 2B-74. A nickel conical gasket is used as a seal at the connections.

(4) Air Systems

The air systems used on the STJ227 engine include various signal lines, labyrinth seal air tubes, aerodynamic brake actuator lines, the turbopump air supply tube, and the secondary air bypass tube system (Boeing installation only). For the Lockheed installation, provision is made for the fuel heater air supply.

Air signal lines provide burner static pressure, compressor discharge pressure, and ambient air pressure. Main burner pressure is routed from a boss located on the burner case to the main fuel control. Another line is routed from a boss on the burner case to both the exhaust nozzle control valve and the afterburner fuel control.

Labyrinth seal air, used in all the bearing compartments, is taken from the compressor fourth stage. This air is routed forward to a strut on the compressor inlet case for the number 1 bearing compartment, and aft to a diffuser case strut for the number 2 and 3 bearing compartments. These lines are Inconel (PWA 1060) tubing, and employ bolted flanges, gold-nickel brazed to the tubes per PWA Specification 19.

The two pneumatic aerodynamic brake actuators, located at the compressor exit, are supplied from an airframe-supplied system. Tube material is Inconel (PWA 1060) with integral ferrules and modified AN connectors (Figures 2B-73 and 2B-74).

The turbopump air supply tube provides compressor discharge air from an air bleed pad located on the diffuser duct to the turbopump inlet case. Bolted angle gasket connectors are gold-nickel brazed to an Inconel (PWA 1060) tube per PWA Specification 19. A universal-type slip joint is employed near the tube center point to allow for thermal expansion. This joint is sealed with metal rings and is similar to that used on the STF219 air bleed tubes.

Secondary air bypass tubes are provided for the Boeing engine version. Ram air is picked up at the engine inlet through eight ducts and routed aft to the ejector section of the engine. A bellows expansion joint, shown in Figure 2B-80, compensates for engine thermal growth. The ducts are fabricated from Hastelloy X (AMS 5536) and are bolted directly to the engine without additional sealing provisions.

A fuel heater is provided in the Lockheed engine version. A tube having a bolted flange with an angle gasket seal draws air from a pad on the diffuser case to the heater.

(5) Plumbing Heatshielding

To prevent heat transfer from the engine cases to the plumbing, a highly reflective gold coating is applied to the tube surfaces. This coating is a compound consisting of metallic gold and an organic resin. It is applied by spraying, then subjected to a bake cycle which decomposes and vaporizes the organic resin leaving a thin coating of metallic gold deposited on the tube. On the JT11D-20 engine, most of

the heat transferred to the tubes from the engine cases is by radiation. A gold coating of 0.00012 inch thickness is used on the tubes to reduce radiant heat transfer; a coating of this thickness will produce emissivities in the 0.03 to 0.05 range.

(6) Tube Joints (seals and fittings)

The modified AN threaded connector tube joints have ferrules machined integral with the tubes from metal that is gathered by a hot upset process at the tube end. This type of joint was developed to replace the conventional silver-brazed joints in the high pressure plumbing and is now used on the JT11D-20 engine.

The following table shows the approximate fatigue strengths of comparable size integral ferrules and brazed ferrules at test conditions of 450°F and 4500 psi internal pressures.

| Sizes (in.) | | Approximate Fatigue Strength (psi) | |
|---|---|---|---|
| Tube | Ferrule | Silver-Brazed Ferrule | Integral Ferrule |
| 0.375 | 0.375 | 22,500 | 32,500 |
| 0.500 | 0.500 | 20,000 to 22,500 | 25,000 |
| 0.750 | 0.750 | 20,000 to 22,500 | 25,000 |

Analysis of test results revealed that variations in the fatigue strength of brazed ferrules can be attributed to braze fit, fillet size, braze coverage, and brazing temperature. Because of these variables, it is difficult to predict the fatigue strength of a particular brazed joint. The fatigue strength of the integral ferrule is affected only by the physical configuration of the machined ferrule and the parent tube material. Although the fatigue strength of the integral ferrule is only slightly higher than that of some brazed joints, the integral ferrule has higher reliability due to the strict quality control that can be exercised over the only variables, upsetting and machining.

On the modified AN threaded connector tube joint sealing is accomplished by the use of a thin nickel gasket compressed between the ferrule and the male connector. This joint and seal configuration is used for low and high pressure plumbing not subjected to thermal shock.

The bolted flange joint (Figure 2B-77) is used for low and high pressure plumbing subjected to moderate thermal shock. Sealing is accomplished by deforming the stainless steel angle gasket into the inner

and outer corners of the seal grooves. This joint is used in the forward section of the afterburner fuel system. The seal grooves are machined into flanges that are gold-nickel brazed to tubes.

Another bolted flange joint (Figure 2B-78) is used for low and high pressure plumbing subjected to high thermal shock. Sealing is accomplished by close control of flatness and parallelism on the sealing surfaces, and by initial flange clamping force to seat the Inconel X (AMS 5542) spring gasket. The unbalanced pressure design increases the sealing force with higher pressures. This joint is used in the aft section of the afterburner fuel system, with seal grooves machined into flanges that are gold-nickel brazed to tubes, and at the afterburner sprayring bosses.

The threaded adapter joint (shown in Figure 2B-73) is also used for low and high pressure plumbing not subject to thermal shock. Sealing is accomplished by deflecting the thin flexible lips of the Inconcel X (AMS 5542) gasket against the conical seat in the boss and the flat surface on the adapter. This type of joint is used on engine components, pumps, controls, etc.

The selection of mechanical tube joints, rather than fittings brazed at engine assembly, is a direct result of experience with the JT11D-20 engine, which employed induction brazed connections in the early stages of development. Several hazards are present when attempting to braze plumbing on the engine. First, a joint with no braze appears the same as a fully brazed joint on X-ray examination. Second, unless extremely careful fixturing is employed, parts close to the brazed joint may become overheated and damaged. In addition, the ability to correct for leaks or to remove engine accessories is so laborious and time consuming that engine development and maintenance are severely hampered. The mechanical joints have been developed to the point of being trouble-free even in high Mach number environments.

(7) Tube Supports

Tube supports are designed to allow for tube thermal growth, to eliminate tube vibration, and to maintain clearance between the tubes and engine or other components. Standard MS loop clamps (AMS 5510) are used to clamp stand-offs to the brackets. Typical bracket assemblies are shown in Figures 2B-70 and 2B-71.

The sliding brackets are used to allow for thermal growth. A slot in the brackets permits the tube to slide in the direction that minimizes stress as the engine thermal gradients increase. The brackets are positioned to provide adequate tube support and low stress. Static (fixed) brackets are used at support points that require no movement due to thermal growth. These brackets support the tubes and eliminate vibration. A combination of sliding and static brackets can be used to shift the load from a critical area to a non-critical area of a given tube. The bushings are assembled in the bracket slots with the retainers and are flared as shown in Figure 2B-70. This provides foolproof bracket assemblies for convenient installation and maintenance.

The same computer program used to select a suitable tube route is also used to locate the brackets. This program reports the stress at any point in a tube with any desired combination of support points, and also calculates the required direction and length of the slots in the brackets.

All tube brackets are made of Inconel X (AMS 5542). When fully heat-treated, these brackets have useful strength up to 1500°F and good oxidation and corrosion resistance up to 1800°F. The retainers and bushings shown are made from cobalt base alloys. They have good oxidation, thermal shock, and corrosion resistance up to 1800°F. The retainers are cast from Stellite 31 (AMS 5382) to reduce manufacturing cost and the bushings are machined from L-605 (AMS 5759). The areas of the brackets that are in contact with the sliding retainers and bushings are coated with diffused aluminum (PWA 44). Laboratory tests and extensive JT11D-20 experience have proved this combination of metals and coating to have excellent wear and strength properties at temperatures equal to those on the STJ227.

Vibration dampers are used on high pressure hydraulic tubes to reduce vibratory stresses transferred to the tubes by a combination of hydraulic fluctuations and mechanical vibrations. The damping is provided by friction between the damper and the bracket (Figure 2B-70). The friction is controlled by the normal load of the Belleville washers in the damper housing. This load is of a magnitude that does not interfere with tube movement due to thermals. The damper housings are made of L-605 (AMS 5759) with Waspaloy (AMS 5709) Belleville washers. They have been used successfully on the JT11D-20 engine at temperatures and vibration levels comparable to the STJ227.

All tube support points use mechanically attached stand-offs, which consist of collars and sleeves and eliminate any chafing or wear on the tube. A clamp stand-off schematic is given in Figure 2B-81. The sleeves are supported by collars similar to a simply supported beam, allowing the sleeves to absorb vibration by flexing between the supports. The sleeves are made in three segments of 120° each. This arrangement reduces wear by providing maximum contact area between collars and sleeve segments, without imposing undue manufacturing controls on these parts. The three-piece-segments also equalize the stiffness of the tubes about their axes at the stand-off locations.

The sleeves and collars are made of L-605 (AMS 5537 and AMS 5759) to obtain the necessary strength and wear properties at elevated temperatures. The corrugated sleeve sections are manufactured by stamping, and are light and very durable. The collars are brazed to stainless steel (PWA 770) tubes per AMS 2666 (silver braze), and high temperature Inconel (PWA 1060) tubes per PWA Specification 19 (gold-nickel braze). The sleeve sections are lockwired as shown in Figure 2B-81 for easy installation and disassembly.

This variety and flexibility of tube support methods have been used extensively on the JT11D-20 engines. The temperature environment and stress levels used on the JT11D-20 engine are comparable to the STJ227.

## 12. MAIN GEARBOX ACCESSORY DRIVE

The engine accessory gearbox provides drive capability for all engine components and additional aircraft accessory drives as agreed to by airframe companies and Pratt & Whitney Aircraft.

The design of the gearbox relies heavily on the experience gained on the JT11D-20 gearbox, since the basic operating requirements are quite similar. The STJ227 main gearbox is shown in Figure 2B-82.

The gearbox is driven through a set of bevel gears and is mounted on the side of the engine with the towershaft located on the engine horizontal centerline. Loose spline couplings are located on both ends of the towershaft to compensate for deflection and misalignment. The hydraulic pumps are all located on the aft side of the gearbox while the main fuel pump, tachometer, and main oil pump are

located on the forward side of the engine. The main fuel pump and tachometer are placed above the engine centerline, with the main oil pump and scavenge oil pump below the engine centerline.

A gear type scavenge pump is installed as a complete unit in the bottom of the gearbox. It picks up oil from the gearbox and the 2 and 3 bearing compartments and returns it to the oil tank. The gearbox breather system incorporates a high speed centrifugal de-oiler in conjunction with a breather pressure regulating valve. These prevent loss of oil overboard, while maintaining a given pressure within the gearbox and main bearing compartment. The main fuel control is driven through the main fuel pump. The gearbox is contoured to the engine cases to place the components as close to the engine as possible. The arrangement chosen minimizes fuel plumbing, prevents the routing of major fuel lines across the bottom of the engine where they would be a fire hazard in the event of a wheels-up landing, and allows for unobstructed removal of the fuel nozzles from the burner case.

The gearbox is attached to the engine case by three spherical ball-jointed links. One mount point is at the rear end of the towershaft housing, one is on the forward end of the housing above the engine centerline, and one is on the forward end of the housing below the engine centerline. This method of mounting allows the towershaft and stub gearshaft to seek their own initial alignment.

All high temperature gears and shafting are constructed of AMS 6260 material with carburized and hardened teeth and wear surfaces. Gears are of integral shaft design where possible and are straddle-mounted in the housing through anti-friction ball and roller bearings. The designs ensure that harmful resonant speeds are outside the operating ranges.

Past experience with the JT11D-20 gears indicates that the use of a 22-1/2° PA involute tooth form provides very satisfactory power transmission at a gear size compatible with the accessory spacing requirements.

All gears are of balanced tooth strength design, and center distance tolerances are minimized to maintain a large contact ratio to ensure good gear life. The following maximum allowable Hertz stresses are used: 90,000 psi under continuous load, 120,000 psi under overload condition, and 140,000 psi under starting conditions. The maximum allowable fatigue stress in bending is 35,000 psi.

for bevel gears and 55,000 psi for spur gears. These data are based on a maximum oil temperature of 425°F. In addition, shaft deflection is minimized to allow bearing misalignment capabilities to be used for any out-of-position due to manufacturing tolerances.

All shafts are straddle-mounted on anti-friction bearings, capable of supporting both radial and axial loads as required by the application and designed for a life expectancy of 10,000 hours. All bearing races, balls, and rollers are of high temperature steel (PWA 724) and stabilized at 1000°F for minimum distortion. All bearings utilize a steel (AMS 6415) cage.

All rotating races are tight and have positive retention on the shaft or housing. All outer races ride in liners, with a loose bearing-to-liner fit. Thrust bearings are provided on all shafts and provision for bi-directional thrust is made.

The STF227 cast housing construction is designed to achieve the maximum rigidity and minimum location tolerance buildup. The housing is composed primarily of a cast body that provides the front bearing supports, and a cast cover that provides the rear bearing supports. These are fastened together with pins and studs for proper alignment of all gearshafts. The cast gearbox is connected to the towershaft through a main powershaft housed in a stamped and welded sheet metal cylinder. The long main powershaft is provided because the components are located forward of the towershaft pad to allow for the removal of the fuel nozzles for inspection.

Since the housing temperature is a maximum of 440°F, the material selected is AMS 4445, a cast, low density, magnesium-thorium combination similar to that used successfully by Pratt & Whitney Aircraft in the JT3, JT4 and JT8 gearboxes. Stresses in the housing are low and uniform with this type of construction.

All bearing liners are tight in the housing and pinned to prevent movement. The liner material is low alloy steel (AMS 6322) for compatibility with the bearing material.

The accessory drive gearbox is sealed effectively at accessory pads and drive shaft entry points by bellows type carbon face seals. Cartridge type seals are used for easy installation, removal, and replacement. Utilizing JT11D-20 experience, these seals incorporate vibration dampening devices, which eliminate bellows fatigue. The carbon face material selected is Purbon P-2003. This was chosen for its low coefficient of friction at elevated temperatures and for its oxidation resistance.

The carbon lip is of the anti-coke configuration. This design prevents seal lift-off due to polymerization of lubricant on a surface adjoining the seal face. The carbon seal face has a surface finish equivalent to 4 AA, per PWA Specification 351, and is lapped flat within 3 helium light bands.

Bellows material selection is based on the anticipated temperature environment of the bellows. AM 350 (AMS 5548) welded bellows are used because of high vendor experience with this material. The bellows have a heatset requirement to assure minimum relaxation of the bellows material at operating conditions.

The seal plate face has a maximum runout of 0.0005 F.I.R. This surface is hardfaced with chromium carbide, finished to 5 AA (per PWA Specification 351, and lapped flat within 2 helium light bands. The edges of the seal plate are chamfered to reduce edge chipping.

At the more severe sealing locations, additional cooling oil flow is provided either through or behind the seal plate. Seal plates are integral with gearshafts in most locations to reduce weight and to simplify configuration.

Some of the design considerations which have proven to be critical in the development of the JT11D-20 gearbox are presented below. These considerations were adhered to in the design of the STJ227 accessory gearbox.

- Gear webs were sized on the basis of maximum bending moment imposed by extremes in gear mesh.

- Bearings are set as far apart as possible to minimize the misalignment due to tolerance stackup that creates undesirable gear mesh conditions and seriously reduces bearing life.

- Particular attention is given to the vibration characteristics of lockwashers, tiebolts and gear rims. All effort is made during initial design to prevent resonance within the operating range, especially at the lower nodal frequencies. Development testing will confirm the resonance characteristics of all parts.

- Sufficient oil for lubrication and cooling is supplied to all gears, bearings, and seals. In particular, seals and bearings are constantly bathed with fresh oil to prevent buildup of coked oil, which

can disrupt the seal lip contact and cause abrasive material to be distributed throughout the lubrication system.

- All splined gears and couplings are provided with two concentric pilots to maintain concentricity and to eliminate bending loads from being transmitted across a spline.

## 13. CONTROL SYSTEM

### a. Introduction

The control system for the STJ227 engine is a hydromechanical system that is essentially the same as the control system on the JT11D-20 engine with modifications only as required to meet specific SST airframe requirements and the increased flow requirements of the STJ227 engine. This control system has been satisfactorily developed on the JT11D-20 engine, and no major new problems are anticipated in the SST application. Component bench and engine qualification tests were successfully completed with fuel inlet temperature and ambient temperatures higher than the SST application.

The control system includes an engine-driven positive displacement main fuel pump; a hydromechanical main fuel control and exhaust nozzle area control; a windmill bypass, low idle, shut-off and drain valve; a hydromechanical afterburner fuel control; an air turbine driven centrifugal afterburner pump; afterburner manifold drain valves; and a high pressure engine hydraulic system utilizing engine fuel and including an engine-driven hydraulic pump and suitable hydraulic actuators for positioning starting bleed doors, compressor inlet guide vanes, exhaust nozzle flaps, thrust reverser flaps, and a spark ignition system for the main burner and afterburner. The control components are described in this section. Installation details are illustrated in installation drawings elsewhere in this report.

### b. Control System Features

The following control system features, which are supplementary to the JT11D-20 control system, are required for the STJ227 engine:

(1) Control Levers

One control lever is used to modulate thrust from maximum afterburner to full reverse; one cuts off main engine fuel flow and schedules low idle.

Power lever operation is shown in Figure 2B-83. A two-position thrust reverser is provided, thereby requiring a cut off lever because fuel cutoff from idle is required. Null thrust can be furnished as an optional feature, with power lever modulation combined with the fuel cutoff lever similar to the arrangement proposed in Phase I.

(2) Fuel Filter

A ten-micron fuel filter is added to the main fuel pump between the centrifugal stage and the gear stage. This filter also provides an electrical signal indicating an excessive filter pressure drop to alert the pilot to energize the fuel heaters if high pressure drop due to fuel icing is encountered. Optional provisions for fuel de-ice heating by a pilot-actuated compressor discharge air-to-fuel heat exchanger are provided.

(3) Ignition System

An electrical spark ignition system is provided for both the main burner and the afterburner. The JT11D-20 engine incorporates a chemical ignition system utilizing the injection of pyrophoric fluid to ignite both burners. This system operates satisfactorily in all respects on the JT11D-20 engine and is available as a developed backup for the SST application. However, an electrical spark ignition system similar to other Pratt & Whitney Aircraft commercial engines was selected for the STJ227.

(4) Fuel-Oil Cooler and Bypass Valves

A fuel-oil cooler and suitable temperature sensing fuel bypass valves are provided in the afterburner fuel system and the main burner fuel system to cool oil and prevent excessive fuel temperatures with resultant coke formation. When fuel temperature approaches the maximum limits, fuel is bypassed to the aircraft tank. A complete analysis of engine heat rejection is stated in Section III-17.

(5) Optional Functions

The following optional functions can be provided if detail studies show they are required: Low idle provisions in the main fuel control to conserve fuel on descent, provision to reset engine rpm as a function of Mach number or shock position to optimize aircraft inlet airflow at high Mach numbers, and an aircraft approach velocity control to hold aircraft approach velocity constant with limited authority.

(6) Afterburner System

A two-zone afterburner system is provided by adding another pressure regulating valve and metering valve to the afterburner fuel control to handle the afterburner fuel flow modulation required for the SST. The afterburner control scheduling cams can be contoured to provide a reduced fuel/air ratio at high $T_{t2}$ to provide optimum thrust-drag relationship as an option if required for the installation.

(7) Component Housings

Aluminum housings for components are provided where permitted by the reduced fuel and ambient temperatures of the SST.

c. Description of Control System

The control system for the STJ227 engine schedules engine operation as a function of power lever position such that thrust variations are essentially linear with power lever position, and such that air flows are compatible with the air induction system over the range of the flight envelope. The control system provides rapid thrust modulation of the engine while ensuring safe, reliable operation during the transients.

The relationship between power lever and engine thrust from full reverse to maximum afterburner is shown in Figure 2B-83. Maximum thrust is obtained by moving the power lever to the full forward position. At this position the turbine inlet temperature is set at its maximum value, and the afterburner provides maximum thrust. When cruise conditions are reached, the power lever is repositioned to the afterburner modulation range. In this range the turbine inlet temperature is set at a reduced cruise value, and the desired thrust is obtained by afterburner modulation.

Engine speed is controlled by modulating the exhaust nozzle area. Open exhaust nozzle area is maintained for idle and until the constant governing speed is reached, thereby providing minimum idle thrust. Low power-to-full afterburner thrust settings are accomplished at constant rotor speed biased by compressor inlet temperature, thereby providing essentially constant airflow for a particular flight condition. This schedule is particularly compatible with high Mach number aircraft air induction systems. The main burner fuel schedule is illustrated in Figure 2B-84 which shows a representative acceleration and deceleration schedule. The governor droop lines in this curve are steep in the open exhaust nozzle region to

control engine speed, and are flat in the constant speed exhaust nozzle area governing range where slope provides essentially constant turbine inlet temperature. The overspeed governor droop line is also shown.

The afterburner fuel schedule is shown in Figure 2B-85. Two afterburner zones are provided with separate pressure regulating valves and throttle valves in the afterburner control to provide accurate fuel scheduling over the desired afterburner modulation range. Zone I is initiated at minimum afterburner PLA and Zone II at a higher afterburner PLA. The afterburner schedule is biased by compressor inlet temperature.

d. Description of Components

(1) Main Fuel Pump

The engine-driven main fuel pump is a two-stage unit incorporating a centrifugal boost stage and a gear type second stage. The second stage consists of two parallel operating pumps with discharge check valves to permit the remaining gear pump to operate in the event one fails. A relief valve is provided in the discharge from the gear pump to prevent excessive fuel system pressure. A 10-micron self-relieving filter is located between the boost stage and the gear stage. The main fuel pump flow passages are shown schematically in Figure 2B-86.

This pump design is essentially the same as gear pumps provided on other Pratt & Whitney Aircraft engines.

(2) Main Fuel Control, $T_{t2}$ Sensors, and Exhaust Nozzle Control (Refer to Figures 2B-87, 2B-84, and 2B-76.)

The main fuel control is a hydromechanical device designed to:

- Schedule fuel flow to the main engine combustion system from full reverse to maximum afterburner.
- Control engine airflow and rotor speed by regulating exhaust nozzle area.
- Control the position of the engine inlet guide vanes.
- Control the position of the engine compressor start bleed system.
- Provide an arming signal to the afterburner control.

- Provide pressure signals to the windmill bypass, low idle, shutoff, and drain valve assembly.
- Provide the following optional functions if proved necessary: low idle fuel flow, reset of rotor speed as function of Mach number or shock position, constant aircraft approach velocity control.

The control computes fuel flow ratio using power lever position, engine speed, and compressor inlet temperature as the basic parameters. Fuel flow ratio is defined as fuel flow in pounds per hour divided by engine burner pressure in pounds per square inch absolute. This ratio is presented by the axial position of a cam follower on the three-dimensional multiplying cam. Engine burner pressure is sensed by an absolute pressure bellows system. The bellows force is transmitted to a force balance servo, which rotates the multiplying cam. The final position of the multiplying cam follower establishes the position of the metering valve servo and, therefore, the position of the metering valve.

The amount of fuel metered to the engine is directly proportional to metering valve position. This is accomplished by a pressure regulating system that acts to maintain a constant pressure differential across the metering valve. All fuel in excess of that metered to the engine and used to actuate the control servos is returned to the fuel pump interstage. The control servo flow is returned to the pump inlet through a regulating valve that sets a controlled minimum pressure in the control body. The bypass return fuel to pump interstage is returned through a combination proportional and integrating bypass valve. The proportional bypass valve prevents large changes in metering valve pressure drop under transient metering valve area conditions; the integrating bypass valve corrects for small errors in metering valve pressure drop.

The main fuel control servo systems are supplied with a regulated servo pressure, which is maintained at a constant level above servo drain (main body pressure). This results in a constant servo power regardless of fuel pressure level. The main control also supplies the exhaust nozzle control and compressor inlet temperature sensors with regulated servo supply and servo drain pressures.

Compressor inlet temperature is sensed by either one of two gas-filled bulbs that produce a gas pressure level proportional to temperature. The gas pressure acts against a diaphragm, creating a load on a lever balance system opposed by a servo system and bellows. The fuel servo pressure required to balance the system is proportional to the gas pressure and, therefore, is proportional to compressor inlet temperature.

The fuel servo pressure signals from each of the sensors are received by the control, which selects the higher of the two pressures. This pressure is sensed by a bellows and is amplified through a force balance servo to rotate the three-dimensional cluster, plateau, and droop governor cams as a function of compressor inlet temperature.

The speed servo system receives its input signal as a force generated from flyweights, which are rotated at an rpm proportional to the engine rpm. This force signal is amplified by a force balance mechanism to translate the three-dimensional cluster cam so that its position is proportional to the flyweight force. Surge and overtemperature protection for any condition of engine speed and compressor inlet temperature are provided by the acceleration contour of the cluster cam.

Governing rotor speed is scheduled as a function of compressor inlet temperature. The desired speed is represented by a radius on the speed countour of the three-dimensional cluster cam. The cam follower moves a roller pivot point in a force balance system, resulting in a specific servo pressure level which represents actual rotor speed. This signal is supplied to the exhaust nozzle control which is set for a pressure representing 100% governing rotor speed. Any variation from this fixed pressure represents speed error and is amplified through a servo system to operate a slide valve for supplying hydraulic pressure to the exhaust nozzle actuators. The exhaust nozzle control servo system utilizes both a proportional system for fast response to large speed error signals and an integrating system for greater accuracy.

Also incorporated in the governing speed control system is the capability to limit nozzle area to a value less than maximum when in nonafterburning operation and to remove this limit when afterburning is initiated.

A pressure level representing a percentage of governing rotor speed operates a hydraulic switch, providing a pressure signal to arm the afterburner fuel control.

Additional contour cuts on the speed-temperature, three-dimensional cluster cam provide signals to position the starting bleed doors and the inlet guide vanes. The signal is amplified through a servo system to a force that is capable of moving the two high pressure slide valves, supplying hydraulic pressure to the bleed door actuators and the inlet guide vane actuators.

The three-dimensional plateau cam schedules fuel flow ratio as a function of inlet temperature and power lever position during operation at governing speed. Droop schedules or speed governing schedules below governing speeds, are computed and transmitted to the multiplying cam follower as a function of engine speed, power lever position, and compressor inlet temperature through action of a three-dimensional governor cam and a set of flyweights separate from those in the speed servo.

A sequencing pilot valve operated by the power lever cam directs pressure signals to operate the windmill bypass, low idle, shutoff, and drain valve. A similar power lever-operated sequencing valve is used to operate the thrust reverser (not shown on schematic).

The control also prevents excessive pressure differential across the engine burner case by a cutback in the multiplying cam at the limiting burner pressure.

Main pump discharge fuel entering the control passes through a 150-micron main filter. Servo flow is further screened through a 75-micorn filter.

(3) Windmill Bypass, Low Idle, Shutoff, and Drain Valve

This unit provides low idle, shutoff and bypass valves to direct fuel either to all the engine burner nozzles for normal operation, to the low idle nozzles for low idle operation, or to the recirculation circuit for windmill operation in response to signals from the main fuel control. During either normal, low idle, or bypass operation, this unit maintains sufficient fuel system pressure for normal servo operation in the main fuel control and for utilization of fuel flow for cooling the lubricating oil in the main fuel-oil cooler.

A drain valve is provided within this unit to drain fuel from the engine fuel manifold when the engine is shut down.

(4) Afterburner Fuel Pump

The afterburner fuel pump (Figure 2B-88) is a continuous duty, turbine driven, centrifugal fuel pump designed to deliver required afterburner fuel flow and pressure. Compressor discharge pressure air is supplied to the turbine through a butterfly valve. The butterfly valve position is controlled by the afterburner fuel control to drive the afterburner pump at a speed required to provide the necessary fuel pressure. This minimizes the fuel temperature rise and the amount of compressor bleed airflow required. The airflow metered by the butterfly valve passes

into a volute, then through the turbine nozzles and the axial flow turbine blades and is then exhausted through a venturi. The venturi creates a vortex in the turbine discharge air during overspeeds, which imposes a back pressure proportional to the overspeed, thus providing an effective speed limiting device with no moving parts.

The pump impeller incorporates an axial flow fuel inducer to minimize inlet pressure requirement, and a centrifugal impeller. Fuel leaving the impeller flows into a volute and is diffused through an outlet nozzle which terminates at a discharge flange.

Engine fuel is supplied from the turbopump discharge through a fixed orifice to lubricate the ball bearings at both the pump and turbine ends of the shaft and to provide continuous fuel cooling for the rotating shaft seal and bearing compartment. The shaft seal provides fuel-to-air sealing, with an intervening overboard drain.

(5) Afterburner Fuel Control

The afterburner fuel control is a hydromechanical unit which schedules metered fuel flow as a function of power lever position, burner pressure ($P_b$), and compressor inlet temperature ($T_{t2}$). Schematic drawing of this component are shown on Figures 2B-88 and 2B-75.

The fuel metering throttle valve is positioned as a function of main engine burner pressure biased by power lever position and $T_{t2}$. The power lever system establishes the nominal position of the rate bar, and the $P_b$ system establishes a roller position along the rate bar. The output of the power lever cam is biased by a three-dimensional $T_{t2}$ cam. A pilot valve operated servo system provides force amplification and establishes a rate lever angle proportional to the desired power lever system position. The throttle valve is spring loaded against the roller so that throttle valve position is proportional to rate bar angle and roller position. In the minimum fuel flow condition, the throttle valve is fully closed and the minimum fuel flow is established by an adjustable orifice in parallel with the throttle valve.

Burner pressure is measured by a bellows system referenced to zero absolute pressure. The resultant bellows force is utilized within a null balance feedback servo system to establish $P_b$ servo shaft position. The roller position is mechanically established by the $P_b$ servo shaft position.

Compressor inlet temperature is measured by a gas-filled bulb at the engine inlet. The resulting gas pressure is amplified by a servo system to rotate a three-dimensional cam within the $P_b$ servo system.

Throttle valve pressure drop is controlled to a fixed value by a pressure regulating valve in series with the throttle valve. The differential pressure across the throttle valve is used by the pressure regulator sensor to establish the pressure regulating valve position.

The pump controller employs a proportional plus integral servo system to position the butterfly valve in the afterburner fuel pump air supply line to maintain a fixed differential pressure from throttle valve inlet to pressure regulating valve discharge. By controlling this differential pressure to the desired value, sufficient fuel pressure is assured without the pump operating at an excessive speed and discharge pressure. A rupture disk is provided within the pump controller to close the butterfly valve in the event of a failure which would cause excessive pressures within the afterburner fuel system.

Initiation of afterburning requires both the proper power lever position and a predetermined minimum percentage of governing speed. When afterburner operation is initiated, afterburner fuel pump discharge pressure is supplied to the servo side of the recirculating valve to close it, and afterburner pump inlet pressure is supplied to the spring side of the shutoff valve allowing it to open. Afterburner shutoff requires that the power lever be in the nonafterburning region regardless of the engine speed. This supplies afterburner fuel pump inlet pressure to the sequencing valve, which reverses the above pressures to the recirculating valve and shutoff valve, thus causing the shutoff valve to close and the recirculation valve to open. During non-afterburning operation a minimum fuel flow is returned through the recirculating valve to the thermal bypass valve for control and lubrication oil cooling. Snap action is achieved in the sequencing valve through the use of spring-loaded balls and latching detents in the valve bore.

The sequencing valve ports afterburner pump discharge pressure and afterburner fuel control body pressure to a two-position switching valve which supplies operating signal pressures to the afterburner sprayring drain valves. When the afterburner fuel control is in a non-afterburning condition, the switching valve supplies afterburner pump discharge pressure to the sprayring drain valve actuator piston to open the drain valves; when in an afterburning condition, pump discharge pressure is supplied to close the drain valves. In both cases, the switching valve vents the low pressure side of the drain valve actuator piston to the afterburner control body.

The large turndown ratio of the STJ227 engine requires a second zone in the afterburner. A duplicate pressure regulating valve and a three-dimensional cam are required (not shown in the Figure 2B-88). The second zone is selected by the power lever-operated sequencing valve in the afterburner. The afterburner control fuel scheduling cams can be contoured to provide reduced fuel/air ratio at high $T_{t2}$ to provide optimum thrust-drag relationship as an option if required.

A 20-mesh screen located in the control inlet filters the metered fuel flow to protect the unit from foreign matter. A 40-micron screen is also located at the control inlet to filter the servo fuel flow. The 40-micron screen is continually washed by control inlet fuel flow, reducing the possibility of clogging. A relief valve is provided to bypass the 40-micron screen in the event of clogging.

A fuel-oil cooler is located in the Zone I fuel discharge line of the afterburner fuel control, requiring a remote afterburner control shut-off and recirculating valve downstream of the fuel-oil cooler.

(6) Afterburner Manifold Drain Valves

The afterburner manifold drain valves are hydromechanical shear type gate valves with a metal-to-metal gate seal. Two valves are provided, one for each afterburner zone manifold. The gate is opened or closed by an actuator rod connected mechanically to the actuator piston. The actuator piston is pressure operated by signals from the afterburner control to operate the valve simultaneously with the sequencing of the switching valves in the afterburner control.

(7) Hydraulic System

The hydraulic system utilizes engine fuel to provide a high pressure working fluid for control and actuation of the exhaust nozzle, starting bleed doors, inlet guide vanes, and the engine thrust reverser. The system schematic and pump characteristics are presented in Figures 2B-72 and 2B-89.

Fuel is supplied to the hydraulic pump from the main fuel pump boost stage. The engine-driven hydraulic pump is a fixed angle, variable delivery, pressure compensated, piston-type pump which discharges fuel at a high working pressure. Pump discharge fuel passes through a 10-micron self-relieving filter. Hydraulic system loop cooling is provided by high pressure fuel diverted from the hydraulic system to the afterburner fuel control and the windmill bypass valve return lines;

also low pressure fuel from the hydraulic pump case is returned to the main fuel pump boost stage. Fuel supplied from the main fuel pump replaces fuel which leaves the system through cooling lines and drains.

(8) Fuel-Oil Coolers

Two fuel-oil coolers provide cooling for the engine lubricating oil by using engine fuel and afterburner fuel as the heat absorbing fluid. When the temperature of the fuel supplied to the main burners is excessive, fuel is returned to the aircraft tank by a suitable bypass valve.

Oil temperature is controlled by using a minimum fuel flow constantly directed through the cooler core. A pressure relief valve is incorporated to bypass excess fuel around the cooler core during high fuel flows and in the event of clogged fuel passages.

(9) Ignition System

A separate low tension, dual-spark ignition system is provided for both main burner and afterburner ignition. Fuel is used to cool the critical exciter components, thus making is possible to use electrical components similar to those on existing commercial engines.

An automatic restart switch can be provided as optional equipment for the main burners. This switch senses engine burner pressure and actuates the ignition system in the event of an engine flameout.

A more complete discussion of the electrical ignition system and tests of systems under SST environment are reported for the STF219 engine in this report.

(10) Thrust Reverser Control

The position of the thrust reverser and level of reverse thrust are controlled by the power lever as shown in Figure 2B-83. The system operates the same as current subsonic jet reversers. It is anticipated that the control for the system will be incorporated in the main fuel control.

(11) Aerodynamic Brake Control

The aerodynamic brake is provided to reduce engine windmilling rpm during emergency conditions. The brake is actuated by pneumatic actuators pressurized by a solenoid operated valve from the airframe high pressure supply. The electrical signal to the solenoid is provided by a manually operated cockpit switch with suitable interlocks with the main fuel control to prevent inadvertent operation.

(12) Oil Tank

The oil tank is located immediately in front of the main fuel pump and control, and is sized for a seven gallon useable oil capacity with allowance for the oil expansion associated with maximum temperature operation. The design incorporates the necessary breather and anti-siphon connection as well as filler and overflow fittings. Construction is patterned after the JT11D-20 oil tank.

(13) Main Oil Pump

The main oil pump design is predicated on utilizing the design principles incorporated into the JT11D-20 oil pump, with gear and housing changes to increase flow capacity approximately 50 percent, and to optimize the space available. The pump contains the usual filter element and pressure regulating and (cooler) bypass valves. It employs carbon bushings, PWA 724 gears, and a cast housing.

e. Component Arrangement

(1) Introduction

The general arrangement of major components on the STJ227 engine has followed the concept of keeping the bottom quadrant of the engine free of controls, pumps, actuators, and lines that contain either fuel or oil. Exceptions are the fuel dump valves which, together with associated supply manifolds, can be emptied of fuel at the pilot's discretion by initiating afterburner or main engine shutdown action when involved in emergency landing operations. The engine installation arrangements shown in Figures 2B-90 and 2B-91 indicate that most components, including the engine accessory gearbox, are mounted along the sides of the engine.

Another basic consideration for arranging components is to achieve the smallest practical nacelle diameter consistent with the desire to use components of proven design concepts and to make adequate provisions for secondary air bypass passages while maintaining a high degree of component commonality in design and location.

The engine accessory gearbox is driven by a towershaft that is located in the horizontal plane and on the right hand side of the engine. The power take-off and starter pad provisions are located on the top of the STJ227 engine. The Boeing installation requires direct coupling to this pad; the pad drives an angle gearbox for the Lockheed installation. For the Lockheed installation all components are arranged to be compatible with the angle gearbox and a Lockheed-required fuel heater. For the Boeing installation all components are arranged to be compatible with the bypass ducts. With these exceptions, commonality between the engines is maintained.

The eight bypass ducts (Boeing) are each 6.00 inches in diameter, with flattened and curved sections at each end to provide clearance with the airframe and to maintain an efficient aerodynamic flowpath. Provisions have been made for suitable bellows expansion joints and duct supports to accommodate thermal effects and pressure loading.

(2) Main Fuel System

The main fuel pump and control are mounted as a unit on the front of the gearbox just above the engine horizontal axis. Three ball joint links on the pump-control package connect to the engine case to provide a non-redundant mount system. This mount system has been successfully used on the JT11D-20. The fuel pump inlet is located near the top of the engine allowing for direct routing of plumbing between the pump and airframe.

The windmill bypass, low idle, and drain valve is located on the bottom of the engine to provide complete draining of the main fuel manifolds and nozzles. External components were arranged to leave open space directly outboard of each of the 32 fuel nozzles for accessibility.

The exhaust nozzle control used in the fuel/hydraulic system is identical to the unit developed for the JT11D-20 engine. It has been located at the left side of the engine to keep hydraulic system fuel from the lower quadrant.

There are four sets of hydraulic actuators on the STJ227 - the inlet guide vane actuators, the starting bleed system actuators, the exhaust nozzle actuators, and the reverser-ejector actuators. The two inlet guide vane actuators are located 180° apart, the lower one being 45° up from the vertical centerline on the right side of the engine. The three starting bleed door actuators are located approximately 120° apart with the two lower ones located 30° below the engine horizontal centerline on either side of the engine. The eight exhaust nozzle actuators are housed inside the struts at the corners of the ejector structure just aft of the ejector mount ring. The design and installation of the reverser-ejector actuation system is covered in Section III-9.

The two aerodynamic brake actuators are located 180° apart on the engine vertical centerline. These are actuated by an airframe pneumatic power source which is supplied through a main fuel control interlock.

(3) Afterburner Fuel System

The afterburner turbopump and control are located on the left side of the engine. The pump is located in the upper quadrant to facilitate piping from airframe to inlet.

The afterburner control is mounted directly beneath the afterburner pump unit to give a direct mechanical linkage between the control and the turbopump air inlet control valve.

The afterburner fuel manifold check and drain valves are located on the bottom engine centerline with the Zone I valve just aft of the rear mount ring and the Zone II valve just forward of the rear mount ring.

(4) Hydraulic System

The engine hydraulic pump is mounted on the aft side of the gearbox just below the engine horizontal centerline. Drive pads for mounting two airframe hydraulic pumps (Boeing) are also located on this side of the gearbox above the engine horizontal centerline. The location of the airframe hydraulic pumps will facilitate plumbing between the pumps and the airframe.

(5) Lubrication System

The oil tank is located immediately in front of the main fuel pump and control. It is supported by three brackets designed to provide a non-redundant mount system.

The oil pressure pump is mounted low on the front of the gearbox to place the oil inlet in a favorable position relative to the oil tank outlet.

The main and afterburner coolers are located on the right and left side of the engine, respectively. This provides the most direct routing of fuel lines from the controls through the coolers to the drain valves and nozzles. The oil scavenge pump is located at the bottom of the engine accessory gearbox approximately 45° below engine horizontal centerline and returns scavenge oil from the number 2 and 3 bearing compartments and engine accessory gearbox to the oil tank.

(6) Ignition System

Two electrical exciters are installed - one each for the main and afterburner ignition systems. The main exciter is located on the left side of the compressor case approximately 20° above the engine horizontal centerline. The afterburner exciter is located on top of the diffuser case. These locations provide good harness routing both from the airframe and to the igniters.

Two spark igniters are provided for the main combustion chamber and two for the afterburner. The main igniters are located on either side of the engine exactly in line with fuel nozzle bosses approximately 45° below engine horizontal centerline. The two afterburner igniters are installed 180° apart on the engine horizontal centerline just forward of the afterburner nozzles. All igniters can be removed without additional engine disassembly.

## 14. ENGINE MOUNTS

### a. General

The STJ227 mount configuration is similar to the STF219. The basic difference is in the engine diameter and the distance between the front and rear mounts. The variations between the two configurations and between the two airframe contractors are presented in Figures 2B-92 and 2B-93.

A number of engine mount schemes were reviewed with both Lockheed and Boeing during the STF219 Phase II-A study period. Due to the differences in nacelle concepts for each installation, two different mount configurations were tentatively agreed upon. These configurations were thoroughly discussed in Phase II-A (Section 18). Except for minor alterations based on engine diameter and length changes, neither mounting was changed during Phase II-B. The STJ219 mount system was, therefore, scaled down by the engine diameter change to fit the STJ227. The maximum allowable flight loads are given in Figures 2B-94 and 2B-95 for the Lockheed and Boeing installation, respectively.

b. Materials

The STJ227 front mount is titanium (AMS 4926) and the rear mount is Waspaloy (PWA 1004) for both the Boeing and Lockheed installations. The rear mount of the STJ227 is an integral part of the turbine exhaust case (Figure 2B-53) and will experience metal temperatures within the acceptable limits of Waspaloy. The temperature of the front mount for both engines will not exceed 525°F (inlet air temperature). This is well within the temperature limit for titanium.

c. Mount Ring Structure

The structural analysis of the front and rear mount is similar to the stress analysis of reinforced circular cylinders subjected to lateral loads (NACA Tech Note No. 1310). This solution considers the effect of shell stiffening from the adjacent shells and inner structure on the mount ring. It has been experimentally verified by laboratory testing and used in the design and development of the JT11D-20 mount system.

(1) Front Mount

(a) Lockheed

The front mount ring is incorporated in the compressor inlet guide vane assembly. The Lockheed inlet case assembly is shown in Figure 2B-32. This assembly is a weldment consisting of an inner and outer ring connected by rigid radial struts (refer to the Section III-1, Inlet Section for detailed description). Mount loads are applied through a ball joint at a locally reinforced section of the outer ring. These loads are resisted by tangential shearing forces in the adjacent cases and by the radial stiffness of the inner and outer ring assembly. Any distortion in the outer ring is resisted by the adjacent case stiffness (shell stiffening) and through the struts to the inner ring. The inner ring will

be the major contributor to stiffness. The deflection of a ring under radial loads is proportional to the cubed power of the radius. Therefore, the small diameter of the inner ring acts as a very stiff hub, resulting in a considerable weight saving. This type of structure has been utilized very satisfactorily for several basic structures in the JT11D-20 and other Pratt & Whitney Aircraft engines.

(b) Boeing

Airframe requirements for bleeding inlet air from the front of the engine for nozzle secondary air, as defined in Figure 2B-96 necessitated an overhung front mount. Therefore, the inlet case could not be utilized for ring stiffness. This will result in a larger mount ring. Mount loads are applied through a ball joint at one location and a clevis joint at another (Figure 2B-93). These points are locally reinforced. Mount loads are transmitted from the mount ring to the inlet structure by 28 local axial struts. These struts are uniformly distributed around the circumference of the engine. This type of structure is used satisfactorily in the JT11D-20.

(2) Rear Mount

The rear mount ring is located outside of the turbine exhaust vanes for both engine installations (as shown in Figure 2B-53). The turbine exhaust vanes are radially free at the O.D. and will not provide reinforcement to the rear mount. The mount is deflection limited due to turbine blade tip clearance requirements. The mount stiffness, which includes the effect of shell stiffening from the adjacent cases, was sized to limit the relative deflection (radially) between the number 2 turbine blade tip and outer case to 0.025 inch. This deflection criterion is a result of JT11D-20 experience which has a similar structure.

Originally, the ejector-reverser mount ring was used for the main engine mount; however, later studies resulted in a separate mount ring. The primary reason for this relocation was to move the engine support out of the afterburner flameholder area.

The mount loads are applied to the rear mount ring through tangential links (as noted in Figures 2B-92 and 2B-93 for both installations. This system was adapted to minimize ring deflection. Because the support is essentially a free ring, radial loads would result in excessive deflections.

## 15. BLEED AND VENT SYSTEMS

### a. Introduction

The following description of the bleed systems refers to those ports or lines that extract air from the engine for airframe use and for engine components. Internal bleeds used for thrust balance and disk and blade cooling are discussed in detail in their respective sections of the report.

The bleed flow requirements for the SST airframe were based on a composite estimate from both airframe companies obtained through field coordination. Experience gained in the JT11D-20 development program was also used in determining the engine bleed flow requirements.

Two sources for bleed air are provided: ports on the engine outer case bleed air from the fourth stage compressor exit to provide engine starting bleed and to supply pressurization for the engine bearing compartment seals; slots in the leading edge of the eight compressor diffuser struts provide air for cabin pressurization, fuel de-icing, airframe anti-icing, engine inlet guide vane anti-icing, and engine afterburner turbopump power.

In addition to supplying air, the compressor diffuser struts have area available for vent lines for several internal sections of the engine. Seal pressurization supply and vent lines for the number 1, 2, and 3 bearing compartments pass through the struts. Also, the thrust balance chamber aft of the last compressor disk is vented to the outer diameter of the engine case. At this point the discharge air is vented to the nacelle. Figure 2B-97 presents a summary of bleed flow estimates for all supply and vent systems.

### b. Design Features

The selection of the bleed locations is based on obtaining a mechanical design without sacrificing pressure and temperature requirements. In several instances, the bleed locations could have been at intermediate compressor stages, but this involves bleed manifolds and case flanges causing unnecessary complexity.

The airframe and engine bleed systems that utilize diffuser struts are supplied and vented as shown in Figures 2B-98 and 2B-99. This system consists of partitioned struts with leading edge slots and a continuous angular manifold at the inner diameter. Air pickup at the strut leading

edges is located toward the inner ends of the struts to provide a source of clean air for airframe use. The compressor discharge air that is contained within the strut-manifold system is available at the outer engine case flanges.

The diffuser struts also provide passages for supply and vent lines for the number 2 and 3 bearing compartment labyrinth seal pressurization flow, and for vent areas for the seal leakage flow that enters the thrust balance chamber aft of the last stage compressor disk.

Cooling flow is required for the turbine exit guide vanes and the afterburner duct system at turbine inlet temperatures greater than 2000°F. This cooling flow will be bled from the sixth stage compressor location.

The requirements for bearing compartment seal pressurization flow are primarily low temperature and sufficient pressure at all engine flight conditions. Fourth stage compressor air satisfies these requirements, and because it has a plenum available, this bleed point was selected.

Anti-icing bleed extraction from the main diffuser struts is based on accessibility and availability of high temperature air.

The starting bleed system on the engine is quite similar to the JT11D-20 system. The location of the bleed is aft of the fourth stage compressor stator. Three actuators, operated by hydraulic system pressure and controlled by engine speed, open and close the bleed ports.

c. Bleed System Testing and Development Background

The STJ227 bleed system utilizes a design similar to that used on the JT11D-20. Extensive testing of this basic system at flight Mach numbers and altitudes greater than those required for the SST has proved the adequacy of the design with respect to available pressures and temperatures.

Testing of the bleed system on the JT11D-20 at simulated flight conditions using appropriate pressure and temperature instrumentation in the compressor diffusers has allowed determination of the bleed system flow-pressure drop relationship. These test data have been generalized into flow function curves based on main engine diffuser Mach number. Resulting correlation curves can be used to describe the flow relationships from the main engine diffuser through the strut-manifolding system to the outer case bleed ports. This method of analysis is easily applied to the STJ227 bleed system.

Venting of the chamber aft of the last stage compressor disk to control thrust balance has been tested on a JT11D-20 at various flight conditions. These tests have demonstrated the ability to vent air from this chamber through diffuser struts. System testing, in addition to analysis, has demonstrated that control of thrust bearing load by this method is practical.

d. Calculation Procedure

The entire engine bleed system is sized and analyzed with respect to flow area and pressure drop using available computer programs. These programs permit the determination of pressure drop-flow relationships for orifices, seals, and supply and vent lines. Compressible flow relationships (e.g. velocity head loss coefficients, flow with friction and heat addition) are available in these programs.

## 16. THRUST BALANCE

a. Introduction

The term "thrust balance" refers to the algebraic sum of forces that act on rotating parts of the compressor and turbine, and it is the net sum of these forces that determines the load acting on the engine thrust bearing. The load is held within allowable limits by controlling the static pressure of all chambers that are inboard of the main gas stream. In addition to satisfying thrust balance requirements, the airflow rates and air temperatures in these chambers must be compatible with cooling requirements of the compressor and turbine disks and blades.

b. Components of Net Bearing Load

Figure 2B-100 typifies the forces acting on a turbojet rotor. The figure shows a single stage compressor tied to a single stage turbine, with

the shaft supported by a single thrust bearing. Using schematic nomenclature, the net bearing load would be defined by the following equation.

$$\text{Net Load} = \left[P_3A_3 + P_2A_2 + L_c + P_6A_6\right] - \left[P_1A_1 + P_4A_4 + P_5A_5 + L_T\right],$$

where:

P = static pressure of air in the particular chamber
A = projected area on which the pressure acts
L = aerodynamic (P) (A) plus momentum blade loads

c. Design Requirements

The design of the main shaft thrust bearing requires the mean effective load of a magnitude to provide high bearing reliability for the life of the engine. Secondly, the minimum loads should be sufficient to prevent skidding. This occurs when the load on the bearing is too small to generate enough friction between the balls and races to overcome the gyroscopic loads generated by the balls. Extensive testing and bearing dynamic analysis have provided Pratt & Whitney Aircraft with a method of predicting minimum loads.

When a bearing undergoes loads of varying magnitudes, the life must be determined using a mean effective load. Since life is inversely proportional to the cube of the load, a high load can have a considerable effect, even though it acts for only a small portion of the total life.

The mean effective load is integrated throughout a mission cycle. It is determined by taking the cube root of the summation of the different loads multiplied by the number of revolutions, and divided by the total number of revolutions.

$$\text{Mean Effective Load (MEL)} = \sqrt[3]{\frac{F_1m_1 + Fm_1 + \text{- - - -}}{m_1 + m_2 + m_3 + \text{- - -}}}$$

$$\text{MEL} = \sqrt[3]{\frac{\sum F\ m}{\sum m}}$$

where:

F = the load acting during
m revolutions

For turbojet operation, where rotor speeds do not vary widely, it is sufficiently accurate to use a constant rotational speed. Therefore, (m) in the above equation becomes percent of mission time.

A summary of the loads presently estimated for the STJ227 engine is presented below. These loads are based on a B-10 life of 10,000 hours evaluated at 5000 rpm. The maximum load is that which will result in a 150 hour B-10 life.

| | |
|---|---|
| Minimum Skid Load (lb) | 3,750 |
| Maximum Allowable Load (lb) | 48,000 |
| Mean Effective Load (lb) | 12,000 |

If a bearing is lubricated, handled, and mounted correctly, all possible causes of failure are eliminated except one: fatigue of the balls or races. The life of an individual bearing is defined as the number of hours at constant speed and load that the bearing is capable of attaining before evidence of fatigue develops in the race or rolling element. Bearing life varies inversely with speed and with the cube of the applied load. If a group of identical bearings are run under the same conditions (such as load, speed, temperature, etc.), a wide dispersion in their lives will result. Statistical methods are applied to the fatigue failures to obtain a desired level of reliability. The usual level of reliability applied to aircraft bearings is 90%. This means that 90% of the bearings will complete or exceed the calculated length of time under the imposed conditions. This calculated length of time is also referred to as the 10% life (or B-10 life). The B-10 life is then a predicted minimum fatigue life at which 10% of the bearings should fail.

d. Mechanics of Bearing Loading

The criteria for optimum bearing load level is associated with two load extremes. The maximum bearing load determines the life of the bearing, and the minimum load determines the margin of safety relative to bearing skidding. Figure 2B-101 shows the action of loading. The three sets of arrows (with associated signs) show that a bearing may have a positive, negative, or zero load. If a bearing is chosen to be loaded either positively or negatively, and the load decreases toward zero, the bearing is then moving in a load direction toward skidding.

e. Definition of General Design Problems

In the choice of a thrust balance system for the STJ227 engine, the prime objective is the selection of chamber pressure levels over the full flight regime that will result in average load levels that produce a long-life bearing and, at the same time, have an adequate skid load margin. The resulting chamber flow system must be compatible with the specific requirements for compressor and turbine disk and blade cooling. For example, the choice of a particular chamber pressure and temperature characteristic over the entire flight envelope may be favorable for thrust balance, but may force certain disks to be unduly heavy because of creep limitations.

f. Choice of Load Direction

The choice of load direction is not important to the bearing as long as that direction does not change at some flight condition. Actually, load direction change may be tolerated as long as the engine does not dwell at a flight condition near the zero or low level point.

In the STJ227 engine, a negative or rearward loaded bearing was selected after careful study of the turbine cooling requirements and the pressure ratio characteristics of the compressor. To meet the requirements for turbine blade cooling, it is necessary to have a fairly high pressure forward of the first stage turbine disk. This means that a high pressure is acting over a large projected area on the disk, imparting a rearward load to the bearing. If a forward direction was chosen, it would have been difficult to maintain the direction at high altitude conditions where pressure ratio decreases. A rearward loaded bearing is also advantageous at windmill and flight idle conditions, because the natural tendency of the engine is to load the main shaft rearward.

g. Effect of Flight Conditions

Generally, at the outset of a particular thrust balance system study, two particular flight conditions are analyzed to give trends of maximum and minimum loads. These conditions are sea level ram, and maximum Mach number and altitude. Because of the particular climb path associated with the supersonic transport, however, the transonic region had to be investigated because it also resulted in low directional loads.

Variations in bearing loads at different flight conditions are caused by changes in pressure levels in the engine. The load tends to decrease at constant Mach number and increasing altitude. Also, as Mach number increases at a constant altitude, pressures contributing to rearward loading increase, thus tending toward a rearward loaded bearing.

h. Method of Calculating Bearing Loads

Pressures and temperatures are obtained from calculated performance data. Because many internal chambers are connected either in series or in parallel by orifices or seals, the pressures associated with them must be calculated by solving flow equations. The entire solution is accomplished by a computer program. The areas of flow orifices are set initially by estimates, keeping in mind the structural requirements relative to pressure differentials, and the thermal compatibility between parts. Aerodynamic blade loads are determined by computer solution, using compressor and turbine physical areas and pressures along with momentum effects. When initial runs are made on the computer, adjustments such as changes in orifice diameters, seal areas, and seal radii, are made until a satisfactory load balance is obtained for all flight conditions. In general, the thrust balance system is designed based on maximum power performance data. It is then checked with flight idle and windmill performance values to ensure compatibility.

i. Proposed STJ227 Thrust Balance System

The flow system for the SST turbojet engine is shown in Figure 2B-102. A negative-loaded bearing characteristic can be incorporated easily because of the relative difference in pressure loads between two major chambers. These chambers are behind the last stage compressor disk, which loads positively (loads shaft forward), and ahead of the first stage turbine disk, which loads negatively (loads shaft aft). Using these two chambers as a base, and by adjusting seal diameters and orifice areas, an acceptable negative bearing load level can be realized. The pressure in the chamber aft of the last compressor disk will have to be lowered to reduce the forward load in this area to achieve a negatively loaded bearing. Flow from this chamber will be dumped overboard or to the nacelle through the main diffuser struts. Consideration will be given to using this flow for engine case cooling prior to venting overboard. Seal clearances and diameters forward of the first

stage turbine must be sized to minimize leakage into the main stream and to obtain a desirable bearing load range. Also, the turbine sealing arrangements between the turbine disks and aft of the second stage disk should be as required to obtain enough turbine cooling flow for the disks, while maintaining pressure levels that will allow for minimal passage of leakage flow into the turbine main stream. An excessive amount of leakage will tend to decrease the efficiency of the turbine.

j. Testing Methods

Two methods using full scale engine tests, verify calculated thrust loads. Both of these methods have been used successfully in the JT11D-20 development program. The indirect method involves instrumenting an engine with temperature and pressure probes. All internal non-rotating chambers are instrumented for pressure and temperature, including small chambers between adjacent labyrinth seals. The engine is then operated at several flight conditions, and the resulting data are programmed into a computer. Solutions are then obtained for the pressures and temperatures in those chambers which cannot be instrumented. Computer output is then compared with original estimates and correlations are made. This method also yields other useful data, such as the flow rate of air through adjacent labyrinth seals, which can be used to calculate running seal clearances.

The direct method utilizes a calibrated thrust ring that is incorporated in the bearing stackup. This procedure involves the design of the thrust ring which, when adapted to the engine, takes the place of a normally used spacer in the bearing stackup. Prior to engine testing, the thrust ring is calibrated for load and deflection. When installed in the engine, the ring is instrumented with strain-gages and thermocouples for temperature compensation. At each steady-state flight condition, test measurements are made and compared with the calibration. In this way the net load imparted to the bearing is measured directly. At present, the direct method is being successfully used in the JT11D-20 program to develop a thrust load map over certain areas of the flight envelope. A thrust load map is a plot of constant bearing load as a function of Mach number and altitude. This method also yields bearing loads at part power and windmill operation.

## 17. ENGINE HEAT REJECTION

### a. Introduction

A system has been designed that integrates all components of the fuel, hydraulic, and lubrication systems (with connecting plumbing) and satisfies all requirements set forth in the Phase II-A study. These systems are shown schematically in Figure 2B-79.

### b. Basic Ground Rules

In determining the inter-relation of fuel, hydraulic, and lubrication system components the following stipulations and limitations were used:

Fuel - Jet Type A
Maximum fuel temperature (engine) - 250°F
Maximum fuel temperature (main or afterburner manifolds) -325°F
Heat rejection allowed to airframe during cruise - None
Maximum allowable bulk oil temperature (engine) - 425°F
Oil - Type II.

### c. General Description of Fluid Flowpaths

(1) Main Fuel System

Fuel from aircraft boost pumps enters the engine at the main fuel pump inlet and flows through the pump, the main fuel control, the main fuel-oil cooler, and into the windmill bypass low idle shutoff and drain valve. This valve allows fuel to pass through the fuel-oil cooler under engine windmill conditions and thereby maintains oil temperature at an acceptable level. It also schedules the metered fuel flow to the main burner under low idle conditions. By incorporating the second function in this valve, a fuel flow higher than low idle may pass through the oil cooler and return to the modulating thermal return valve through the windmill bypass valve, although the burner flow is much lower. The valve also drains residual fuel from the manifolds when the burners are off.

(2) Afterburner Fuel System

Fuel enters this system at the afterburner fuel pump inlet, and flows through the turbopump into the afterburner fuel control where the flow is metered separately for the two afterburner zones. Zone II fuel flow passes directly from the control to the burner through the drain valve. Zone I fuel passes through the afterburner fuel-oil cooler, the shutoff

and recirculation valve, and then to the burner. The afterburner shutoff and recirculation valve is used to block metered flow to Zone I and to recirculate cooling flow through the afterburner control and fuel-oil cooler when the Zone I burner is not lit.

(3) Hydraulic System

The proposed hydraulic system is an open loop type design using fuel as the actuator fluid. The hydraulic pump inlet flow is taken from the main fuel pump interstage.

Pump discharge flow splits into three flowpaths. The first two are recirculation lines which carry fuel flow, and thereby heat, away from the system maintaining the fuel temperature at an acceptable level. One line is routed to the windmill bypass valve and the second to the afterburner shutoff valve. These are used to cool the recirculation lines during main burning and Zone I afterburning. The third flowpath is hydraulic system actuation flow, which is routed from pump discharge through the main fuel control or pilot valves and then to the high pressure side of the actuator piston. Low pressure return flow from the actuators is joined by the actuator cooling flow lines. This flow is then routed back to the main pump interstage.

(4) Lubrication System

A closed loop system with heat pickup rejected to fuel in the two oil coolers was chosen for the STJ227. From the main oil pump, oil under pressure is passed through the shell side of both coolers. From the main oil cooler discharge, the oil is supplied to each of the three bearing compartments, aircraft power take-off, and main engine gearbox. After performing lubrication and cooling functions, the oil is scavenged to the engine gearbox, through the oil tank, and back to the oil pump inlet.

(5) Fuel System Temperature Management

To hold the maximum fuel temperature within the specified limit, two thermally-actuated valves are incorporated. One of these is a modulating thermal fuel return valve which senses main oil cooler fuel discharge temperature. Fuel returning from the recirculation lines is routed back to the main pump inlet under normal temperature conditions. If the fuel-oil cooler fuel discharge temperature increases to the sensor setting, the modulating valve starts to port some fuel back to the aircraft fuel tanks, thereby decreasing the fuel inlet temperature.

The second valve, a fuel bypass valve which is an integral part of the afterburner fuel-oil cooler, has a fuel temperature sensor located at the cooler fuel discharge. When this fuel reaches the sensor setting, some fuel is bypassed around the cooler thereby lowering the fuel temperature. Under this condition some of the lubrication system heat load would be absorbed by the main fuel-oil cooler.

d. Main, Afterburner, and Hydraulic System Heat Rejection

The heat rejection in the main fuel, afterburner fuel, and hydraulic systems is primarily from pumping, throttling and ambient pickup. The major portion is due to pumping and amounts to approximately 4200 Btu/min. Throttling and ambient pickup account for another 1400 to 1800 Btu/min. Heat rejection varies greatly for different operating conditions, but under the worst conditions fuel system heat rejections will be from 5600 to 6000 Btu/min.

e. Lubrication System Heat Rejection

(1) General

The heat rejection from all lubrication system components to the oil has been calculated for cruise Mach number and altitude. Steady-state oil temperatures have been evaluated and are well within the limits set forth. Oil flow rates have been established for each bearing compartment.

(2) Basic System Operation

Minimum fuel and oil system temperatures during the flight mission and any possible off-design flight condition will be maintained by the fuel-oil heat exchanger in each of the engine heat sinks (main burner and afterburner fuel flows). The maximum expected bulk oil temperature range will be between 330°F and 380°F.

The afterburner-on fuel flows throughout the typical mission are sufficiently high to absorb the total lubrication system heat rejection and the afterburner system pumping and ambient heat pickup. It is unlikely that the fuel bypass valve will bypass fuel flow due to high temperatures during afterburner-on condition unless it is during off-design operation or at severe transient flight conditions.

During afterburner-off, idle, or windmilling, the afterburner system recirculation flow is sized to absorb the full lubrication heat rejection. The afterburner recirculation flow returns to the modulating thermal fuel return valve where it is routed to the engine pump inlets if the main burner manifold fuel temperature is below 325°F. If the main burner becomes thermally saturated, the supplementary cooling valve will split the flow and return a minimum fuel flow and heat back to the aircraft fuel tanks. In bypassing part or all of the recirculation fuel flow to the aircraft tanks, the temperatures in the main, hydraulic, afterburner, and lubrication systems are reduced to acceptable levels.

(3) Detailed Contributions of Lubrication System Heat Load

Figure 2B-103 shows the level of heat input from individual sources and the oil flow rates necessary to transfer the heat while maintaining acceptable oil and component temperatures.

Oil flow rates to the individual bearing compartments were selected commensurate with a 50°F rise in bulk oil temperature. The total oil flow includes that necessary for the jet pump which was sized for primary and secondary flow in a one-to-one ratio.

18. SECONDARY AIRFLOW

a. Introduction

A study has been made of the cooling and heating characteristics of secondary flow that passes between the engine case and engine nacelle walls. The required quantity of secondary airflow is a function of several factors and is different for each engine application. Experience gained from extensive engine testing on the JT11D-20 development program has been applied to the specific requirements of the STJ227.

b. Design Considerations

The guidelines considered in determining the secondary flow, temperature, and pressure requirements commensurate with engine and nacelle integrity are presented below.

- Effect on engine case temperatures and pressure differentials
- Total engine case-to-nacelle heat rejection
- Heat input to controls, bracketing, and plumbing

- Final use of secondary air, such as ejector cooling
- Total temperature rise of secondary air
- Allowable nacelle wall temperature
- Blow-in door, ejector, and reverser design requirements
- Heat input to fuel, hydraulic, and lubrication system fluids
- Back pressure on primary nozzle flaps for hydraulic actuator load calculation
- The heat ratio transferred by radiation to that by free and forced convection.

c. Generalized Nacelle Component and Engine Case Cooling Design Criteria

Engine case temperatures are primarily a function of the working gas temperature within the engine. Even at fairly high secondary flow rates the main stream Reynolds number yields heat transfer coefficients that control wall temperatures. However, as secondary flow decreases to very low rates, the temperature rise of this air over the entire engine length increases to a level where it can influence engine case temperatures and over-ride the influence of low velocities.

The minimum secondary flow is that necessary to give optimum ejector performance and cooling. For the STJ227 this flow is between 2 percent and 3 percent of engine airflow. For 2 percent of engine airflow at cruise flight conditions the air temperature rise is calculated to be 50°F.

The heat transfer rate of nacelle air to plumbing and controls is a function of secondary airflow rate. Several studies performed on the JT11D-20 at a variety of flight conditions have shown that if the plumbing is covered by heatshields, the heat flux (Btu/hr/ft$^2$) is almost independent of secondary flow rates. The airspace between tubes and heatshielding provides a resistance to convective heat transfer, and the shield itself offers a resistance to radiant heat transfer. Also, if a low emissivity coating (such as gold) is applied to the tubes instead of the conventional heatshielding, the heat flux will decrease at very low nacelle flows. However, as nacelle flow increases, a point is reached where the heat flux becomes greater than that which existed with heatshielded tubes. The characteristic trend of heat flux for the

two configurations described is shown in Figure 2B-104. The crossover is a function of flight condition and axial location along the engine. A thermal analysis has been made on a section of the engine case between the main burner and turbine inlet to show the effects of secondary airflow and temperature. Figure 2B-105 shows three arrangements of this section of the engine. These arrangements are: 2 percent secondary flow, a sealed nacelle, and a sealed nacelle with a heatshield between the burner liner and engine case. Figure 2B-106 presents the results of the heat transfer calculations. The engine case temperature increased 25°F with a sealed nacelle relative to the 2 percent configuration. The engine case temperature decreased 225°F when the heatshield was added. These calculations assumed a turbine inlet temperature of 2200°F at cruise conditions. It should be noted that the above differences will vary along the length of the engine. Boeing has tentatively elected to bypass the secondary airflow around the forward portion of the engine by ducts, and Lockheed has elected to flow this air over the engine. Studies will be continued and results modified as specific engine and airframe design criteria become available.

d. Nacelle Wall Heat Transfer Considerations

Secondary airflow has a direct influence on the choice of materials and structural aspects of the airframe nacelle walls. In the "hot" sections of a turbojet engine (turbine and afterburner), a high percentage of heat release from engine cases is by thermal radiation. In applications where a fairly high percentage of secondary airflow is permitted, the heat transferred from engine to nacelle can be carried away by convection, thereby maintaining both walls at fairly low temperature levels. However, as the allowable secondary airflow is decreased, the nacelle walls operate at higher temperatures, and re-radiate heat to the engine cases causing their temperature to increase.

## 19. WEIGHT ANALYSIS

The Phase II-A report presented a weight of 8300 pounds for a 450 lb/sec full afterburning, turbojet engine. Subsequent studies have been concentrated on engines of a 525 lb/sec size. Three air flow schedules were studied (high, base, and low) and were weighed at two turbine inlet temperatures (2000°F and 2300°F). This gives a total of six configurations. The high flow, low temperature engine has a calculated weight of 10,800 pounds and allows a physical overspeed at the Mach 2.7 condition of 1.12 times the sea level take-off rpm. Figure 2B-107 present the weight of this engine by section. Figure 2B-108 shows the weight of the six configurations mentioned above based on the airframe manufacturer's specific installation requirement.

## 20. ASSEMBLY PROCEDURE

### a. Compressor Inlet Case

The variable inlet guide vane of the STJ227 engine is assembled, adjusted, and operated as a separate unit prior to assembly with the remainder of the engine. This ensures ease of removal or maintenance. Individual items that make up the complete assembly are shown in Figure 2B-32.

The pin-bolts, vane flaps, and linkage arms are assembled on the inlet case as a unit to provide the proper angular motion. The flaps are fixtured at the cambered position to eliminate actuation tolerance buildup. The actuators are mounted and attached to the linkage arm synchronizing ring brackets. The fixture is removed and the complete assembly may then be actuated.

The number 1 bearing housing, containing the outer bearing race and rollers, and the rear number 1 bearing support, containing the labyrinth seal lands, are bolted into the inlet case from the rear before assembly with the compressor.

### b. Compressor

The compressor is assembled vertically beginning with the rear compressor hub and the auxiliary disk and moving forward to the front compressor hub. The ninth stage disk is assembled to the rear hub and the auxiliary disk. The flange of the auxiliary disk is slotted to allow the outer tiebolts to be installed. The inner and outer spacers and the eight-to-ninth stage cone are assembled forward of the ninth stage disk. The stator and rotor assemblies are then assembled alternately to complete the compressor assembly. The number 1 bearing compartment seals and the inner bearing race are assembled to the front compressor hub prior to attaching the inlet case.

### c. Diffuser Case

Assembly of the diffuser case starts with the installation of the compressor exit guide vanes and the aerodynamic brake. The towershaft gear and bearing assemblies are next installed in the number 2 bearing housing. The labyrinth seal lands, the carbon seal carrier, and the seal pressurizing and vent lines are then assembled into the number 2 bearing housing.

The outer combustion chamber case is loosely assembled to the diffuser case from the front, and the diffuser case is then installed on the compressor rotor and case assembly.

The seal face and oil scoop are installed on the number 2 bearing sleeve, torqued, locked, and positioned on the compressor rear hub. The main engine thrust bearing is installed in its carrier and placed on the bearing sleeve. The towershaft drive gear is then installed and the stackup torqued and locked. The number 3 bearing support cone is then installed and the lube and seal pressurizing and vent lines are connected.

d. <u>Burner and Turbine Inlet Section</u>

The forward part of the combustor, containing the ram air scoops and swirl cups, is inserted into the diffuser section until machined fittings on the forward face of the combustor are seated on machined platforms in the cutaway trailing edges of the struts. Retaining pins are inserted from outside the diffuser case through the trailing edge of each strut into holes in the outer machined fittings on the combustor. These pins are then torqued and locked.

The inner and outer transition ducts are installed and slid forward over the front part of the combustor. The inner combustion chamber case is bolted to the inner rear diffuser case flange. The turbine inlet guide vanes are inserted into the rear retaining plate lugs and the rub ring and segmented outer front vane support ring are bolted to the outer turbine case. The segmented inner retaining ring is installed and held for assembly purposes by countersunk screws.

The burner inner transition duct is attached to the inner combustion duct and the turbine inlet guide vane inner retaining ring. The outer transition duct is moved rearward and piloted on the forward flange of the turbine case. The outer combustion chamber case is slid to the rear and bolted to both the turbine case forward flange and the diffuser case outer rear flange.

The fuel nozzles are inserted through the diffuser case and bolted. The two igniters are then inserted through mounting flanges on the diffuser case and secured with bolts.

e. Turbine and Number 3 Bearing Assembly

The bearing outer race and support are assembled to the number 3 bearing support cone. The front oil scoop, the bearing inner race and rollers, the rotary seal ring, and the adjacent knife-edge seal ring are assembled on the turbine shaft. The shaft is then pushed forward through the bearing outer race until the spline on the forward end is fully engaged with the mating spline on the compressor rear hub. The threaded tiebolt is then torqued and locked. The number 3 bearing carbon seal assembly is assembled to the outer race support flange. The shaft rear labyrinth seal ring is assembled and the spanner nut is torqued and locked. The heatshield assembly, which contains the seal lands for the labyrinth seals, is assembled to the intermediate flange of the bearing support.

The first stage blades are assembled into the disk with blades of equal moment weight opposite each other, and the front and rear cover plates are bolted into place. After balancing, this assembly is bolted to the front flange of the turbine hub and the tiebolt nuts are torqued and locked. The interstage seal-spacer is bolted to the rear face of the first stage disk. The first stage blade segmented outer shroud is installed in the turbine case. The hub and the first stage disk and blade assembly is then splined to the turbine shaft, and the spanner nut torqued and locked.

The second stage vane inner shroud, vanes, and interstage seal land assembly are installed in the turbine case and locked in position. The second stage blade segmented outer shroud is assembled to the rear flange of the turbine case.

The second stage blades are assembled into the disk with blades of equal moment weight opposite each other, and riveted for axial containment. After balancing, this assembly, together with the rear diaphragm and knife-edge seal ring, are mounted, bolted and locked on the turbine hub rear flange.

f. Turbine Exhaust

The second stage turbine rotor aft knife-edge seal and seal land diaphragm are riveted to the forward flange of the inner turbine exhaust case. The exit guide vanes are bolted to the stiffener rings, and the tailcone and inner radiation shield are bolted to the rear flange. This assembly is held in the outer turbine exhaust case by radial pins.

The turbine exhaust section and the afterburner-ejector system may be assembled to the engine as a unit.

g. Afterburner

The afterburner section of the engine bolts directly to the rear flange of the turbine exhaust case. The flap segments and seals of the primary nozzle are attached to the afterburner rear combustion chamber.

The fuel distribution and flameholder system is installed from the inside and bolted to mounting pads on the diffuser case. The two ignition units are threaded into bosses on the diffuser case.

The fuel baffle is assembled between the diffuser case rear face and the front flange of the combustion chamber case. The segmented liners and heatshields are slid aft into tracks in the combustion chamber cases after the ramp section of the liner has been bracketed inside the rear case.

h. Ejector-Reverser

The hydraulic actuators for the reverser are mounted in the rear of the main ejector support struts with the required plumbing. The blow-in doors and spacer panels are bolted in place between struts. The nozzle synchronizing bellcranks and internal connecting links are installed on the struts.

The entire unit is assembled from the rear over the afterburner combustion duct and connected by eight pins at the ball joint connections to the ejector mount ring. The nozzle segments are connected to internal links and bellcranks. The basic supporting structure is now complete.

The outer shroud is slid forward over the support structure. The reverser door links are set into stop points on struts, and the inner shroud assembly, containing the main structure, liner, and reverser doors, is slid over the main support structure. With the shroud in full reverse position, door links are connected to respective doors. Actuator rod ends and outer shroud are connected to the shroud support system and ring.

The trailing edge flaps are assembled in a loose holding fixture. The flaps are then connected to hinges on the shroud support and ring. After the interconnecting seal plates are installed, the flap holding fixture is removed.

e. Turbine and Number 3 Bearing Assembly

The bearing outer race and support are assembled to the number 3 bearing support cone. The front oil scoop, the bearing inner race and rollers, the rotary seal ring, and the adjacent knife-edge seal ring are assembled on the turbine shaft. The shaft is then pushed forward through the bearing outer race until the spline on the forward end is fully engaged with the mating spline on the compressor rear hub. The threaded tiebolt is then torqued and locked. The number 3 bearing carbon seal assembly is assembled to the outer race support flange. The shaft rear labyrinth seal ring is assembled and the spanner nut is torqued and locked. The heatshield assembly, which contains the seal lands for the labyrinth seals, is assembled to the intermediate flange of the bearing support.

The first stage blades are assembled into the disk with blades of equal moment weight opposite each other, and the front and rear cover plates are bolted into place. After balancing, this assembly is bolted to the front flange of the turbine hub and the tiebolt nuts are torqued and locked. The interstage seal-spacer is bolted to the rear face of the first stage disk. The first stage blade segmented outer shroud is installed in the turbine case. The hub and the first stage disk and blade assembly is then splined to the turbine shaft, and the spanner nut torqued and locked.

The second stage vane inner shroud, vanes, and interstage seal land assembly are installed in the turbine case and locked in position. The second stage blade segmented outer shroud is assembled to the rear flange of the turbine case.

The second stage blades are assembled into the disk with blades of equal moment weight opposite each other, and riveted for axial containment. After balancing, this assembly, together with the rear diaphragm and knife-edge seal ring, are mounted, bolted and locked on the turbine hub rear flange.

f. Turbine Exhaust

The second stage turbine rotor aft knife-edge seal and seal land diaphragm are riveted to the forward flange of the inner turbine exhaust case. The exit guide vanes are bolted to the stiffener rings, and the tailcone and inner radiation shield are bolted to the rear flange. This assembly is held in the outer turbine exhaust case by radial pins.

The turbine exhaust section and the afterburner-ejector system may be assembled to the engine as a unit.

g. Afterburner

The afterburner section of the engine bolts directly to the rear flange of the turbine exhaust case. The flap segments and seals of the primary nozzle are attached to the afterburner rear combustion chamber.

The fuel distribution and flameholder system is installed from the inside and bolted to mounting pads on the diffuser case. The two ignition units are threaded into bosses on the diffuser case.

The fuel baffle is assembled between the diffuser case rear face and the front flange of the combustion chamber case. The segmented liners and heatshields are slid aft into tracks in the combustion chamber cases after the ramp section of the liner has been bracketed inside the rear case.

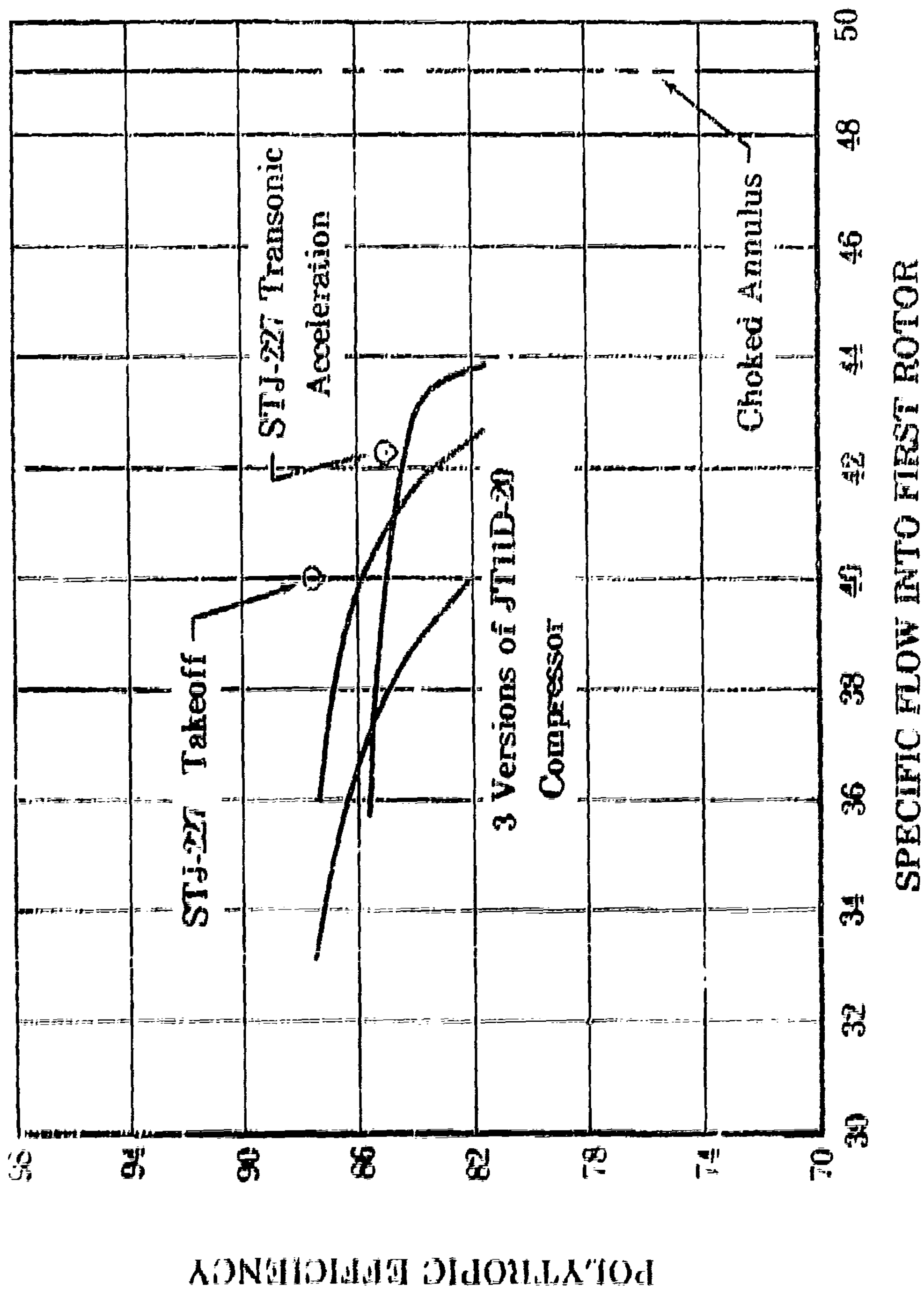
h. Ejector-Reverser

The hydraulic actuators for the reverser are mounted in the rear of the main ejector support struts with the required plumbing. The blow-in doors and spacer panels are bolted in place between struts. The nozzle synchronizing bellcranks and internal connecting links are installed on the struts.

The entire unit is assembled from the rear over the afterburner combustion duct and connected by eight pins at the ball joint connections to the ejector mount ring. The nozzle segments are connected to internal links and bellcranks. The basic supporting structure is now complete.

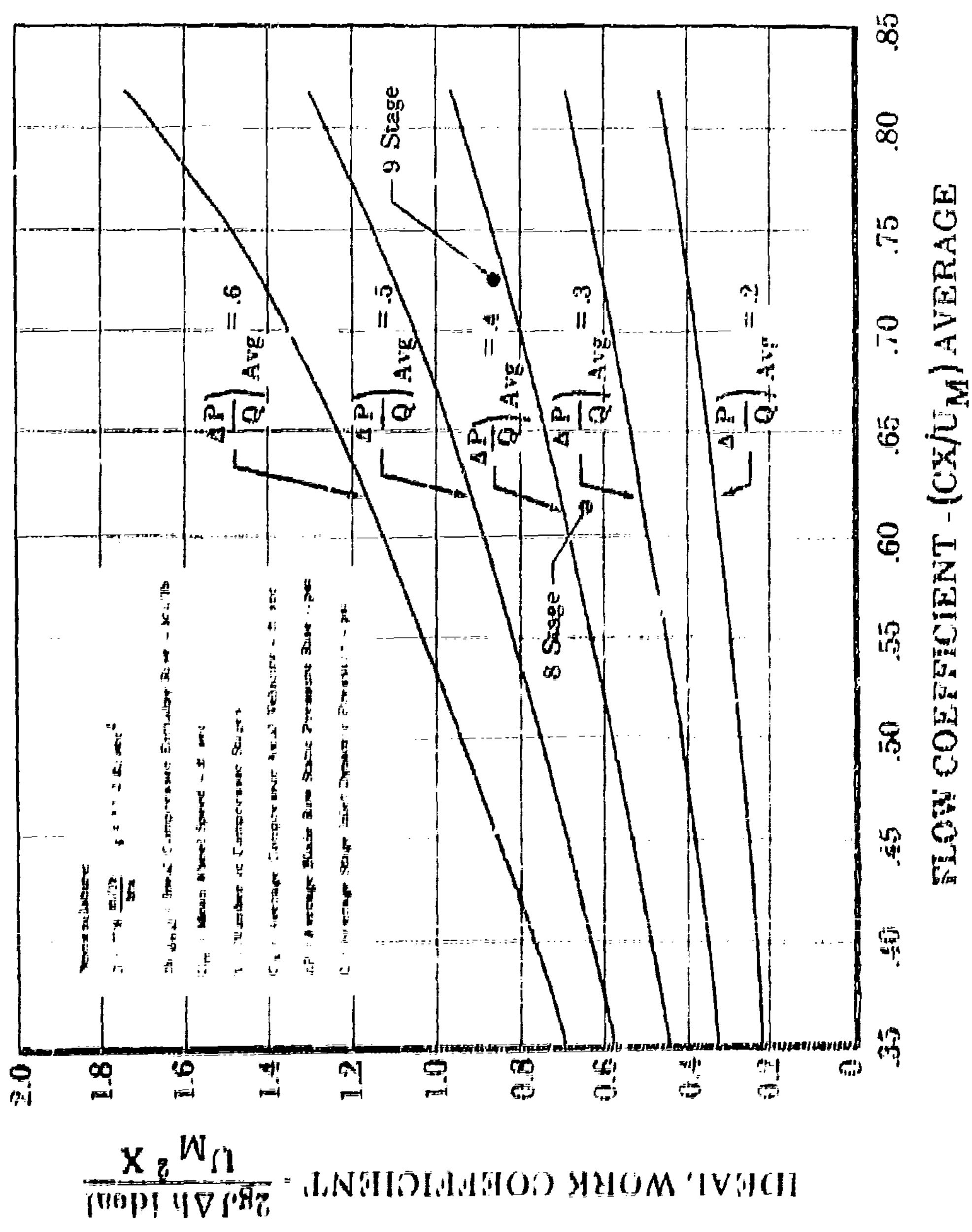
The outer shroud is slid forward over the support structure. The reverser door links are set into stop points on struts, and the inner shroud assembly, containing the main structure, liner, and reverser doors, is slid over the main support structure. With the shroud in full reverse position, door links are connected to respective doors. Actuator rod ends and outer shroud are connected to the shroud support system and ring.

The trailing edge flaps are assembled in a loose holding fixture. The flaps are then connected to hinges on the shroud support and ring. After the interconnecting seal plates are installed, the flap holding fixture is removed.



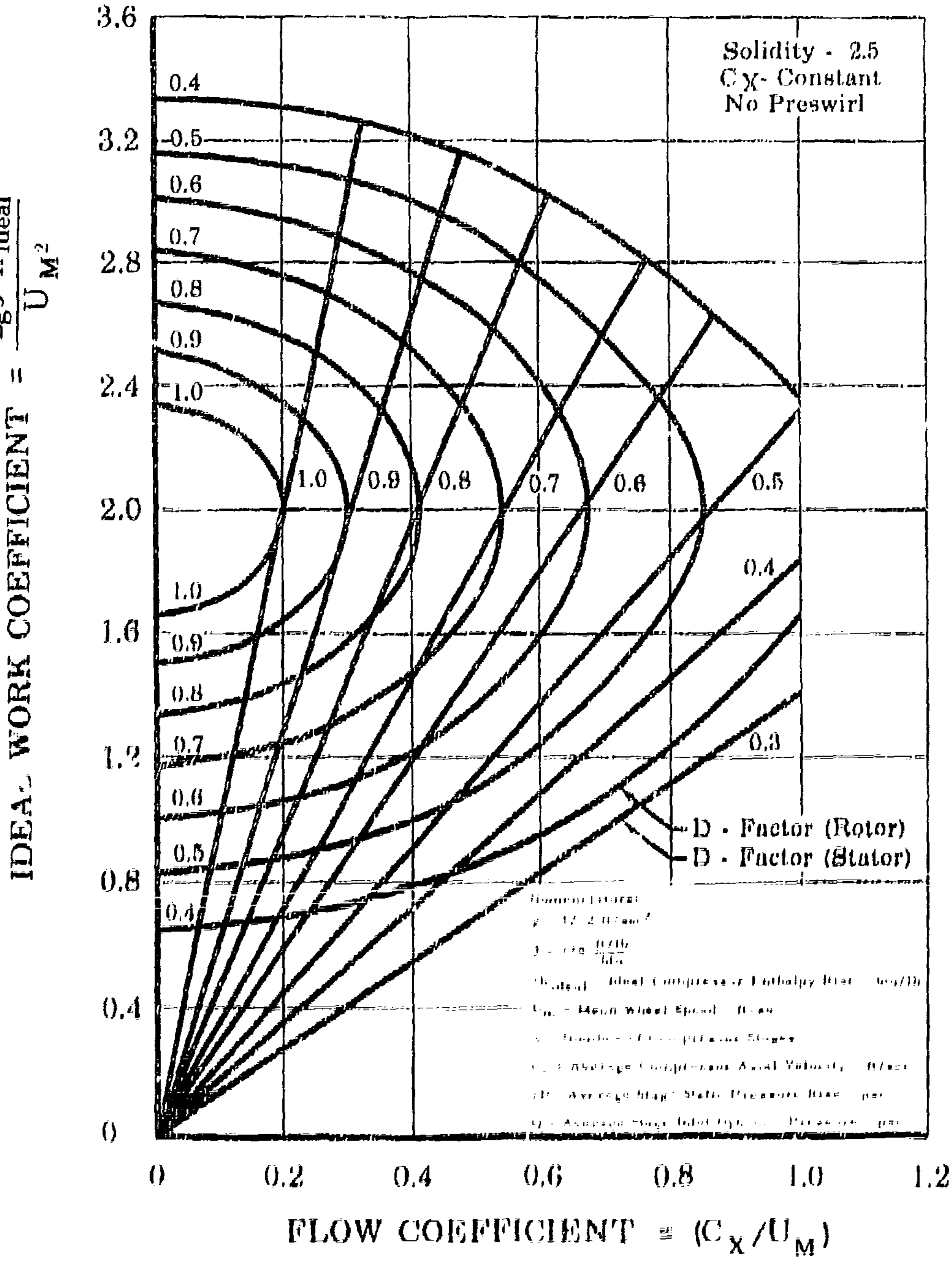

COMPRESSOR POLYTROPIC EFFICIENCY VS. SPECIFIC FLOW

Figure 2B-1



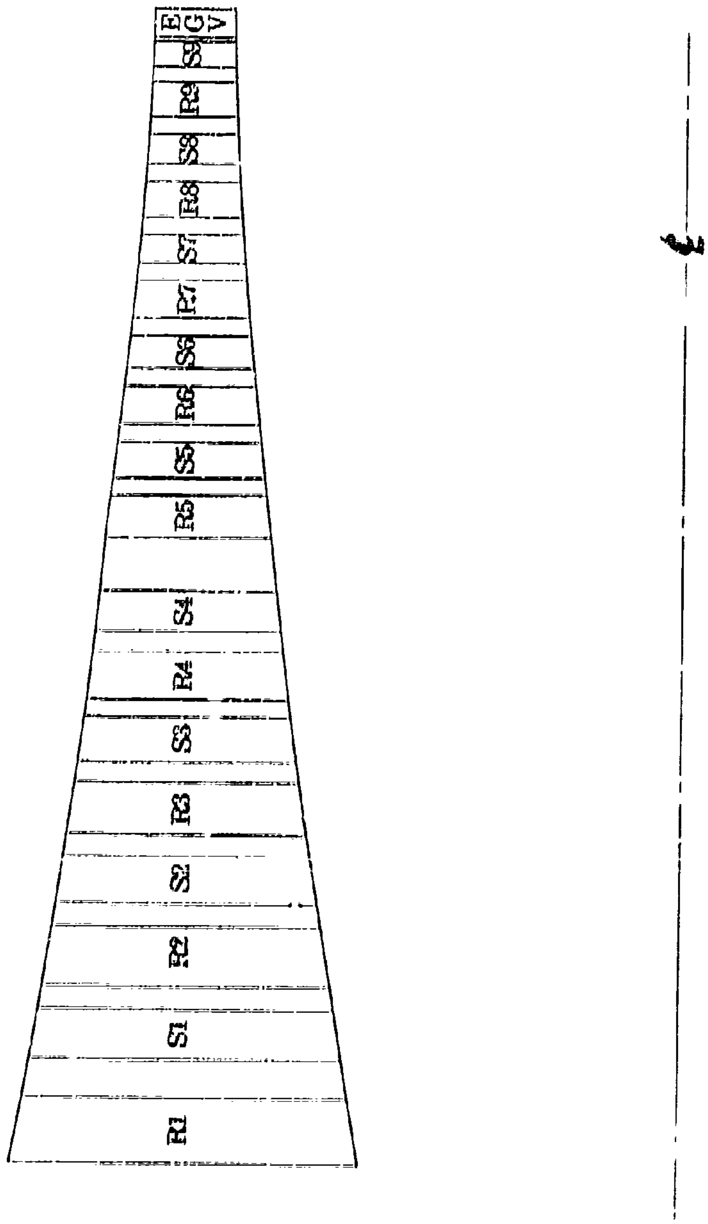

COMPRESSOR PRELIMINARY DESIGN LOADING ESTIMATE

Figure 2B-2



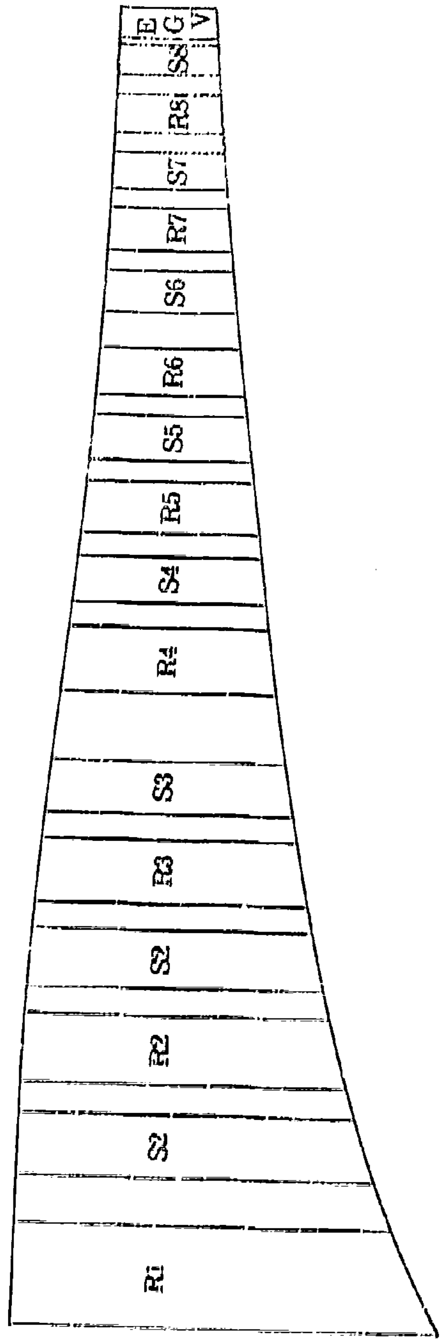

IDEAL WORK COEFFICIENT VS. FLOW COEFFICIENT

Figure 2B-3



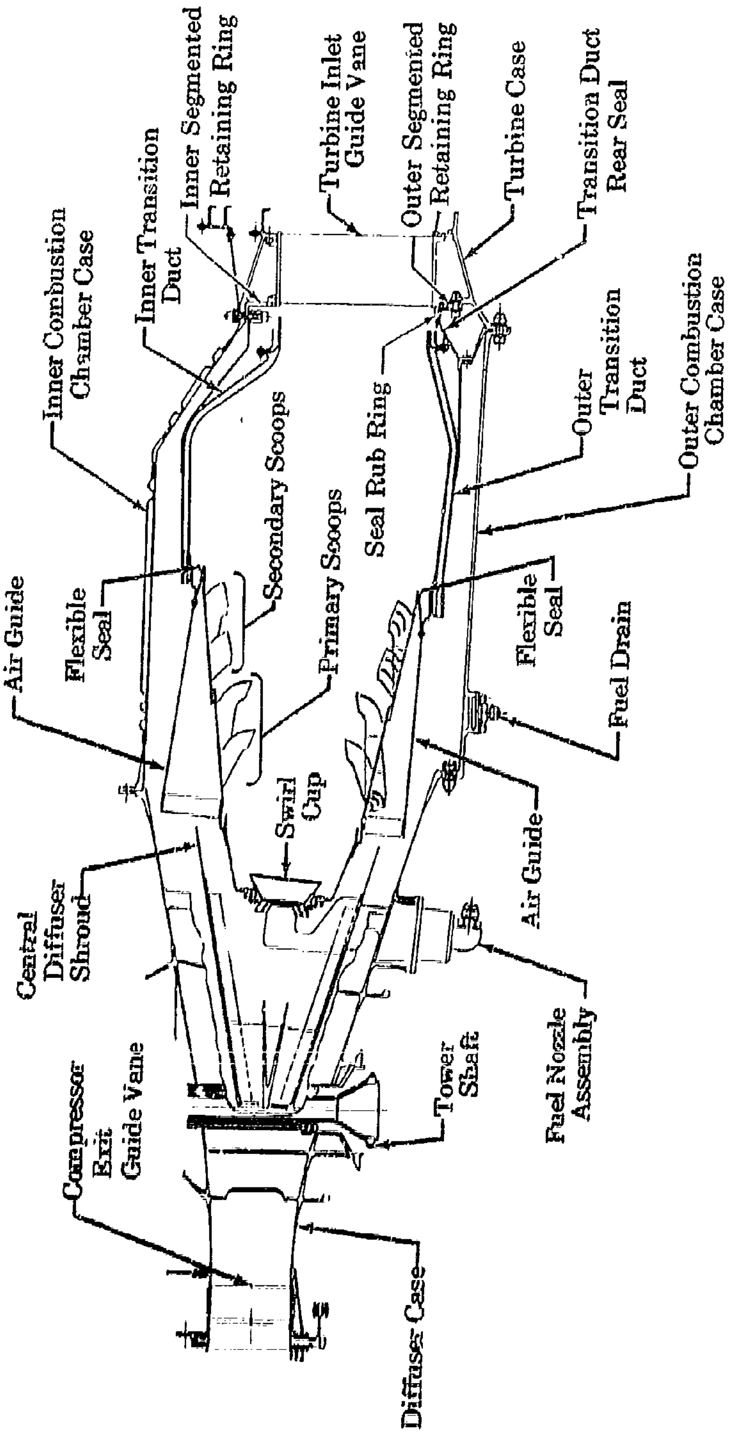

STJ227 NINE STAGE COMPRESSOR

Figure 2B-4



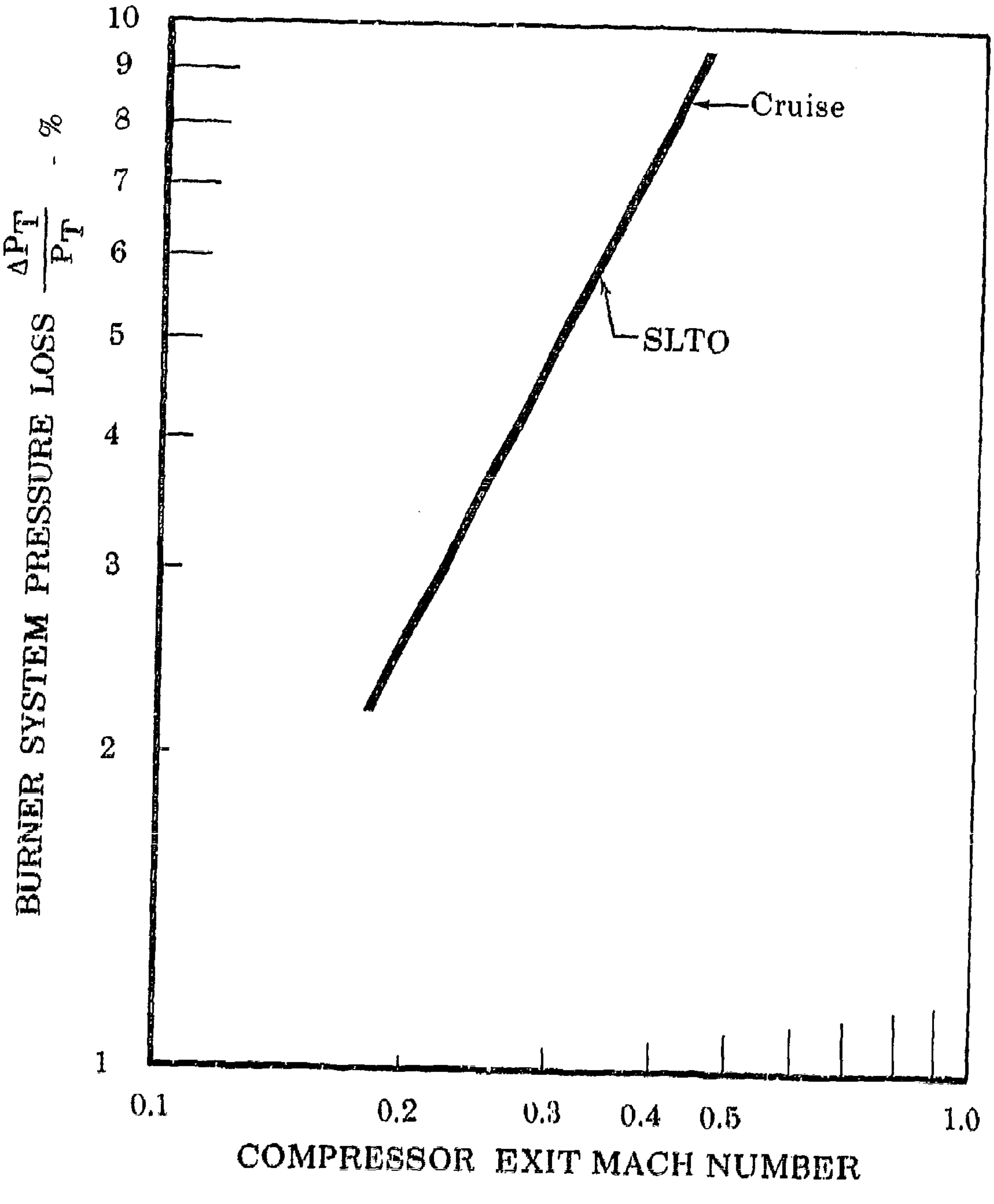

STJ227 EIGHT STAGE COMPRESSOR

Figure 2B-5



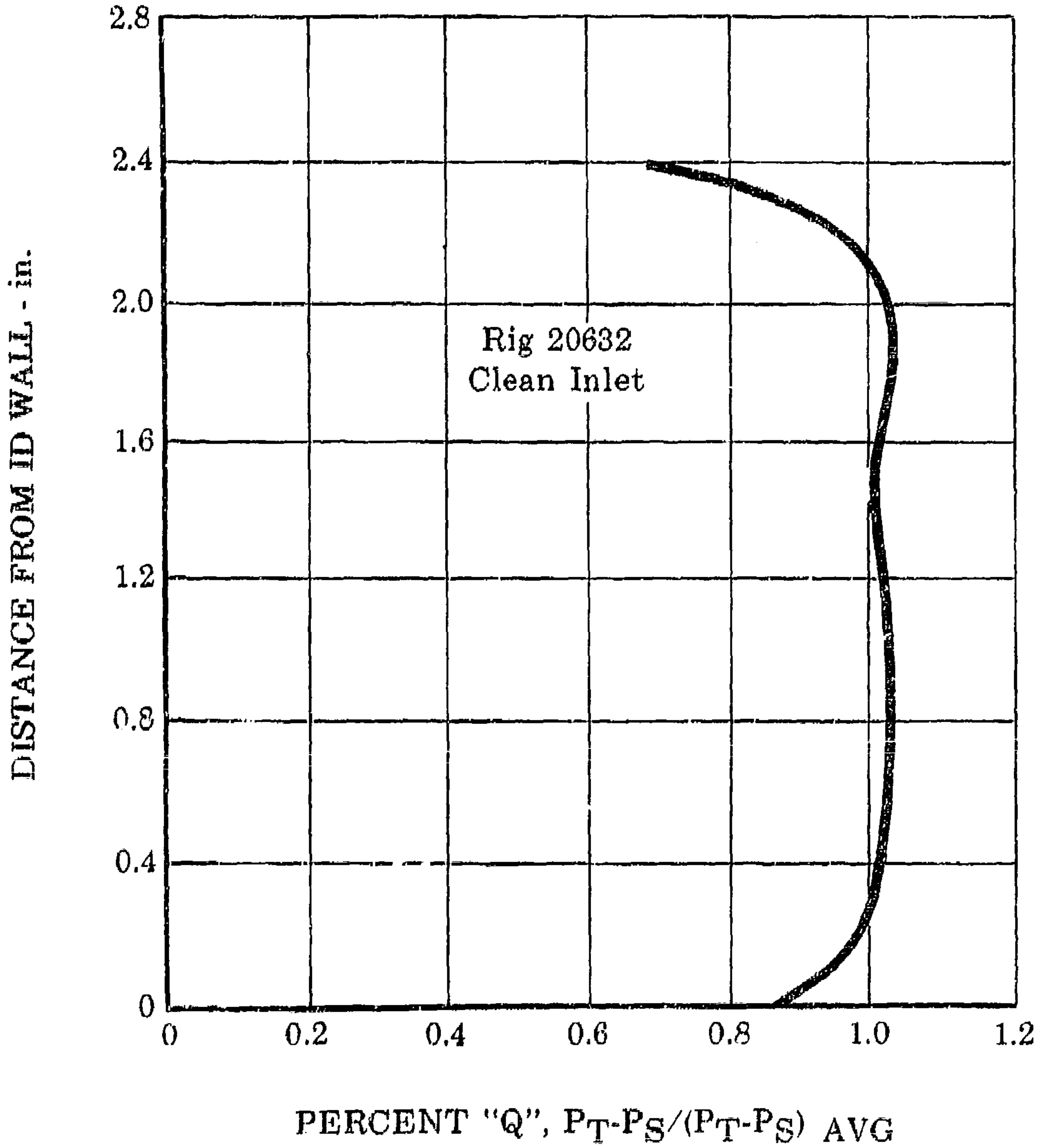

PRIMARY COMBUSTION CHAMBER

Figure 2B-6



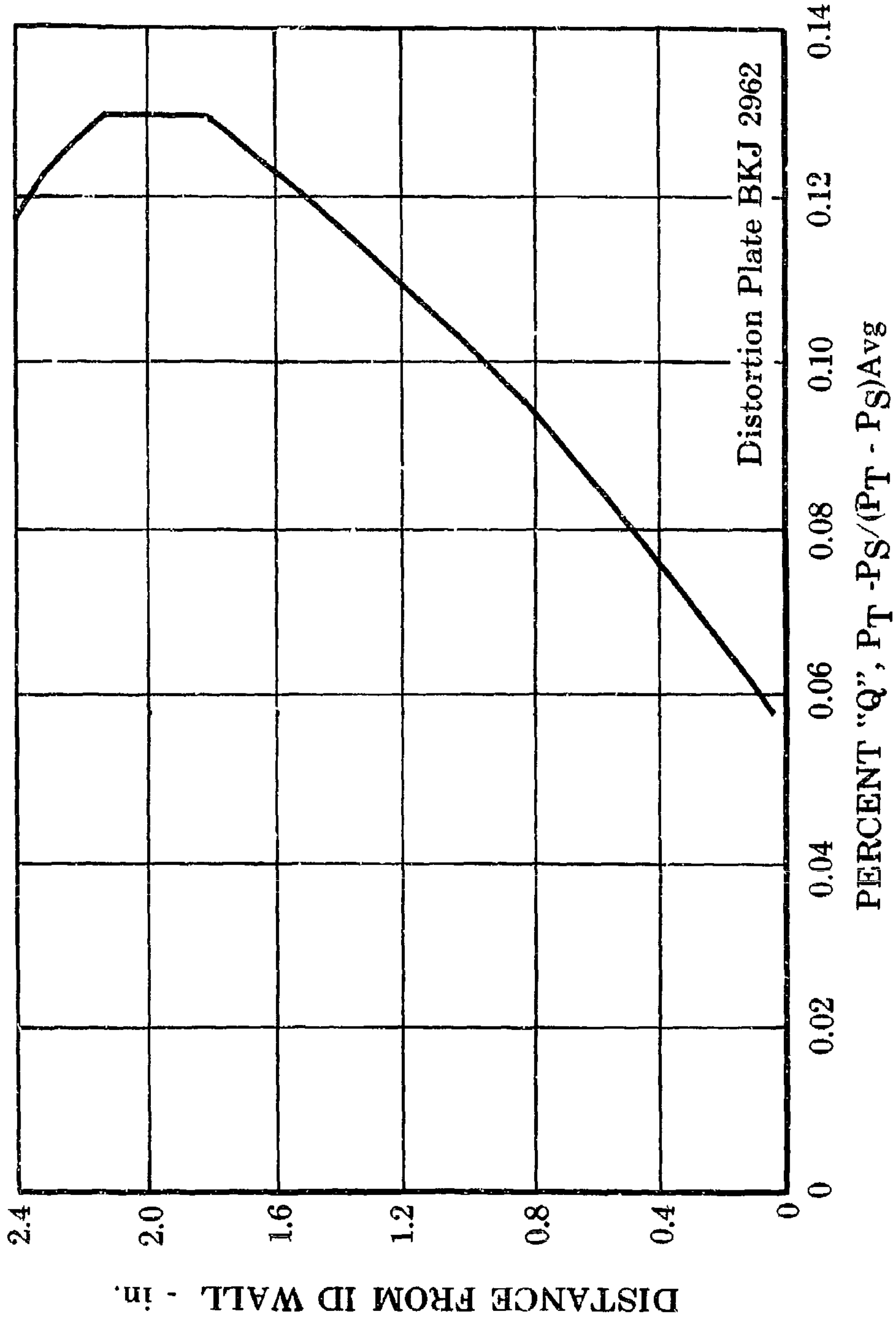

BURNER SYSTEM PRESSURE LOSS ESTIMATES

Figure 2B-7



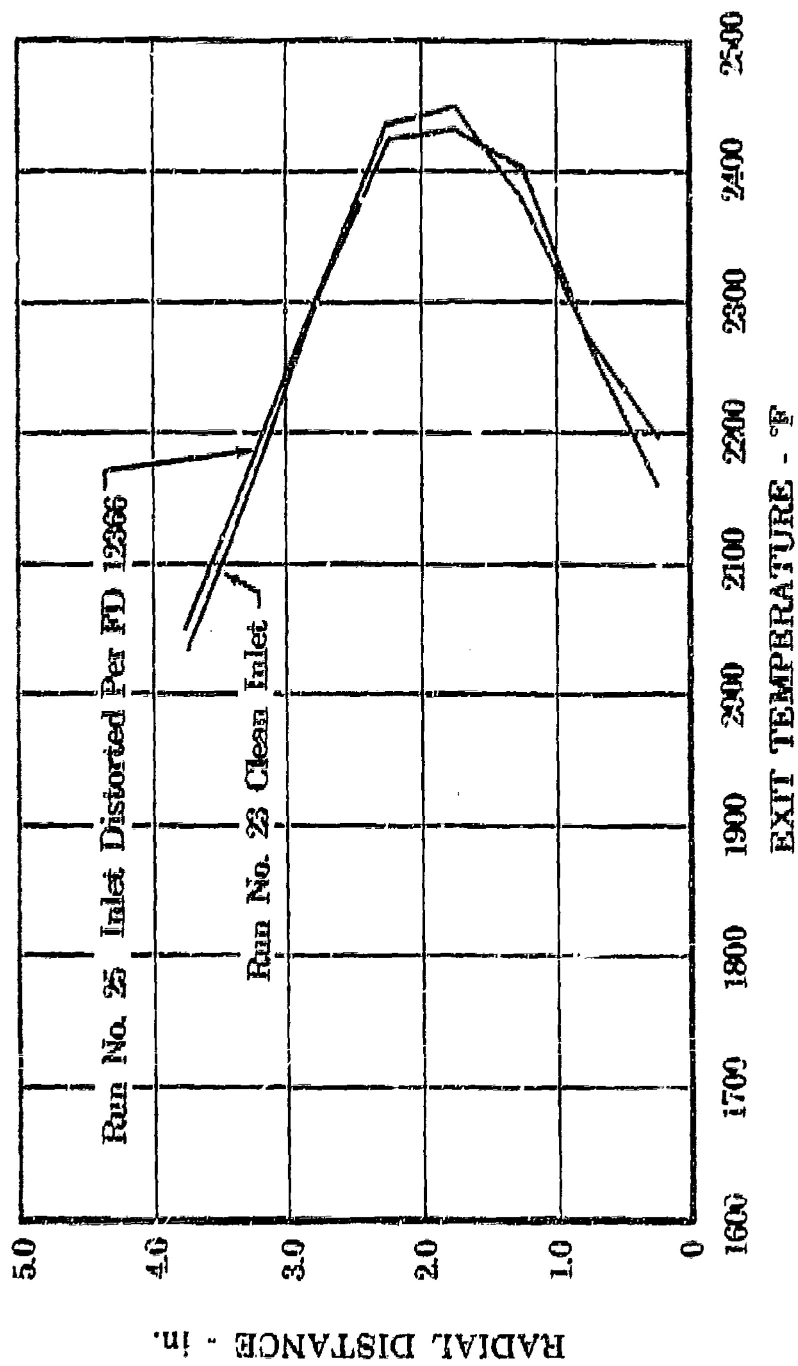

COMPRESSOR EXIT VELOCITY PROFILE

Figure 2B-8



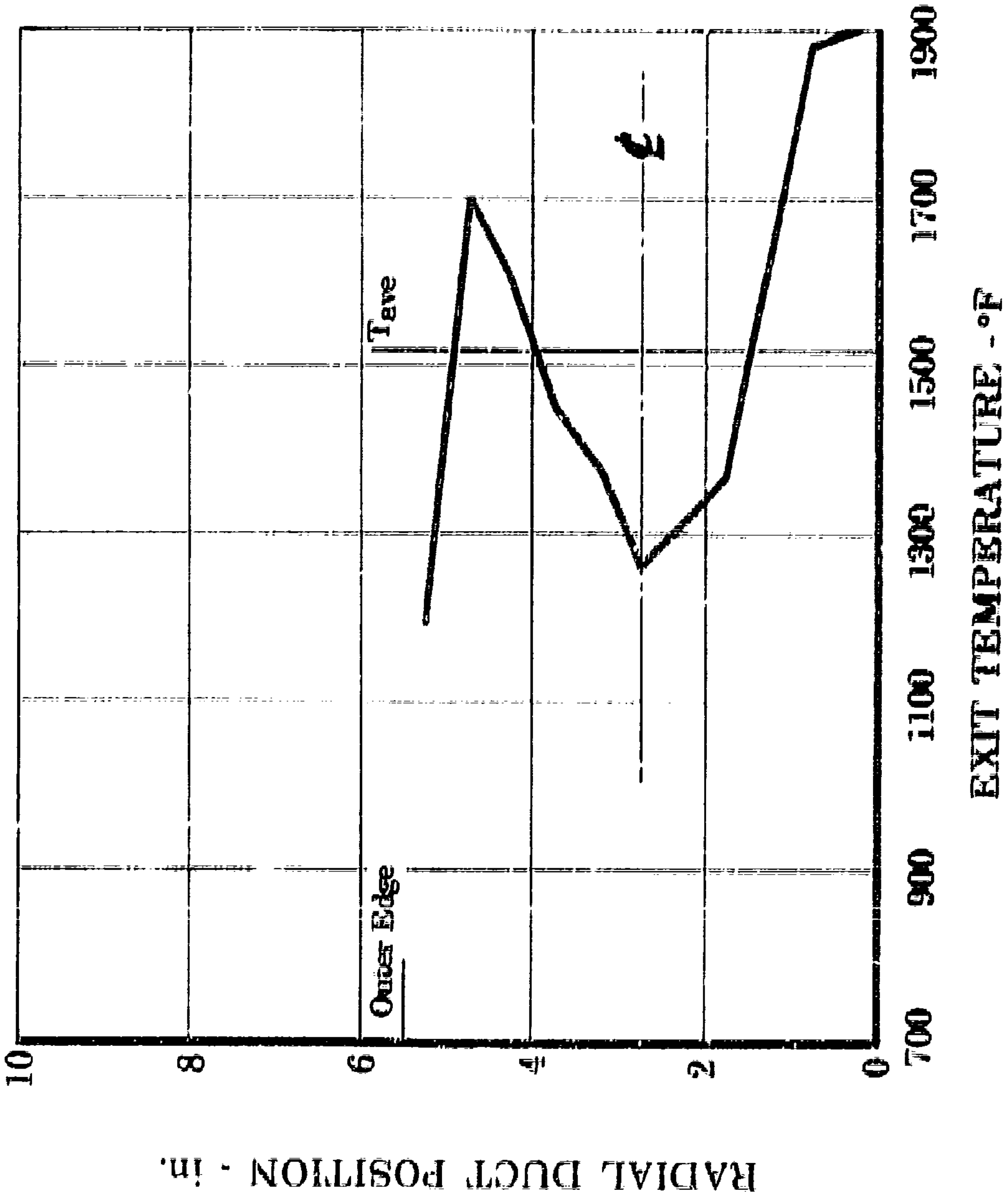

COMPRESSOR EXIT VELOCITY PROFILE

Figure 2B-9



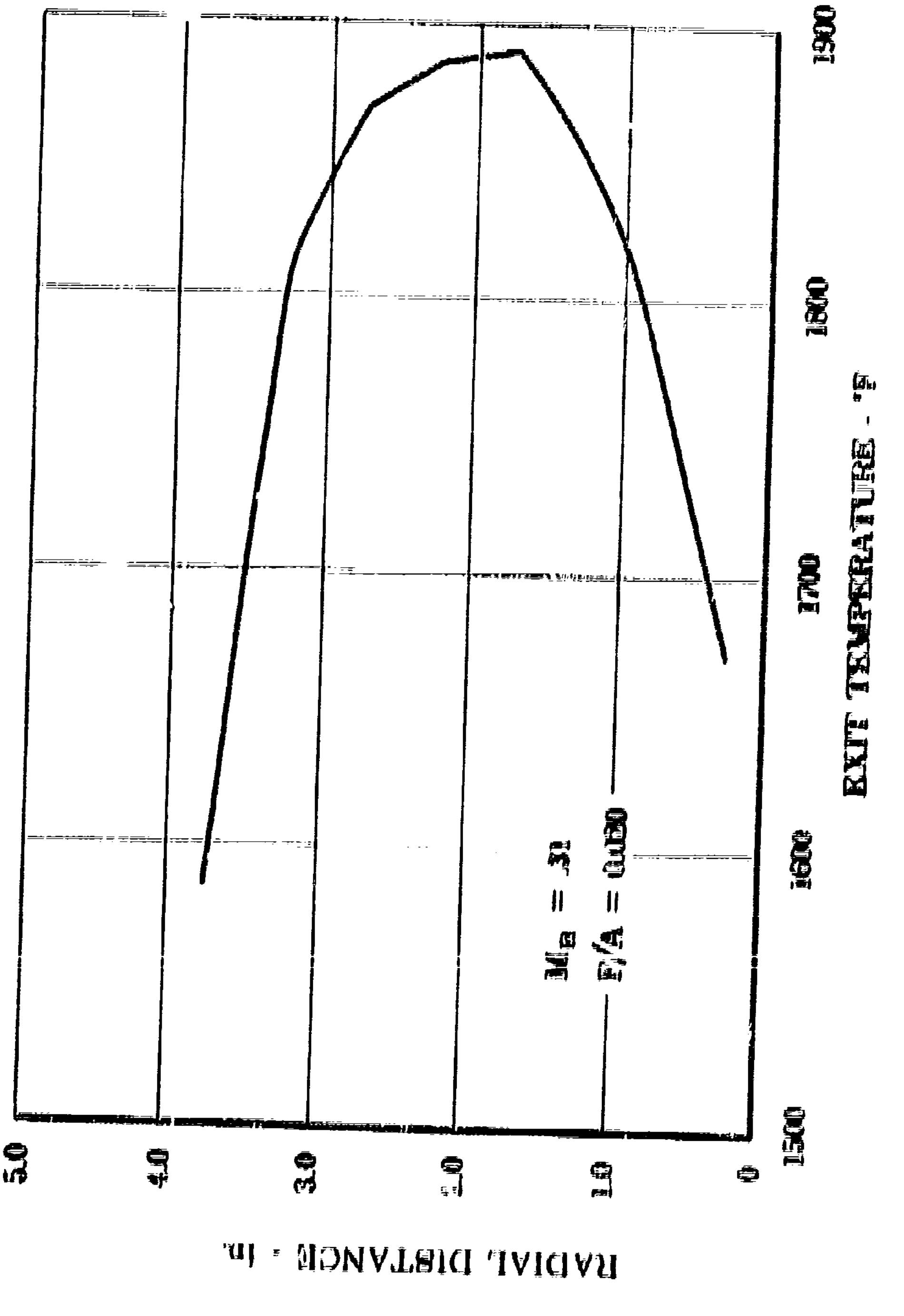

EFFECT OF INLET VELOCITY ON OUTLET TEMPERATURE

Figure 2B-10



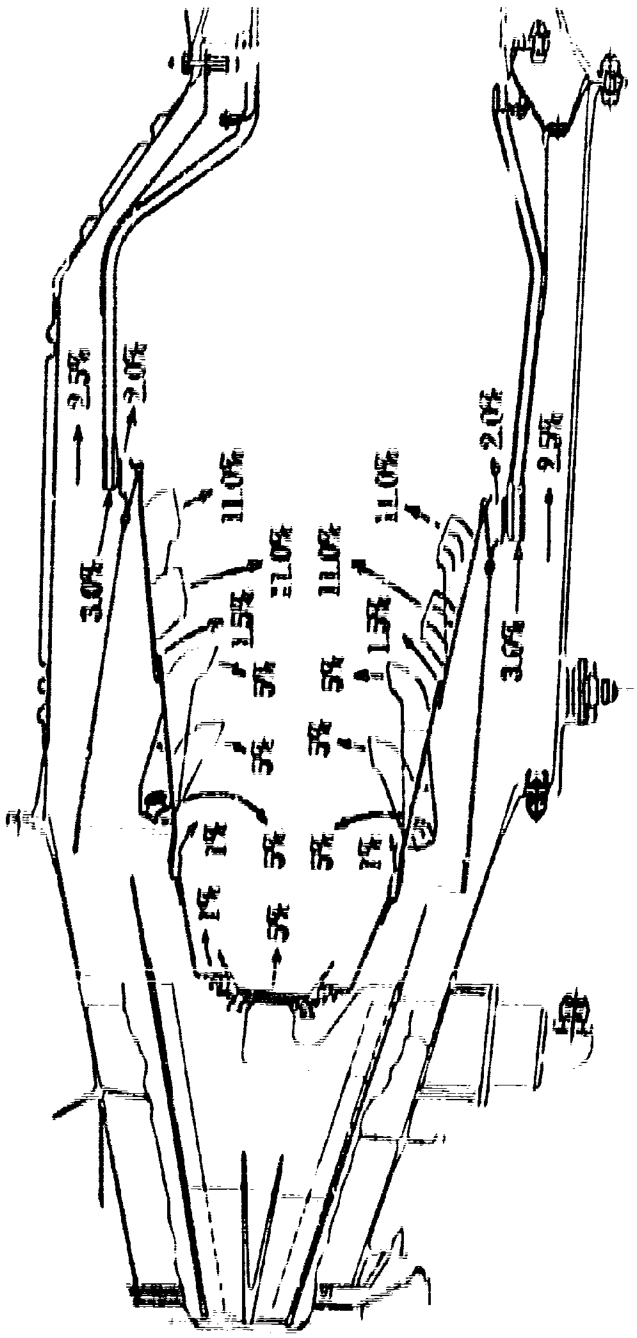

AVERAGE COMBUSTOR EXIT TEMPERATURE PROFILE

Figure 2B-11



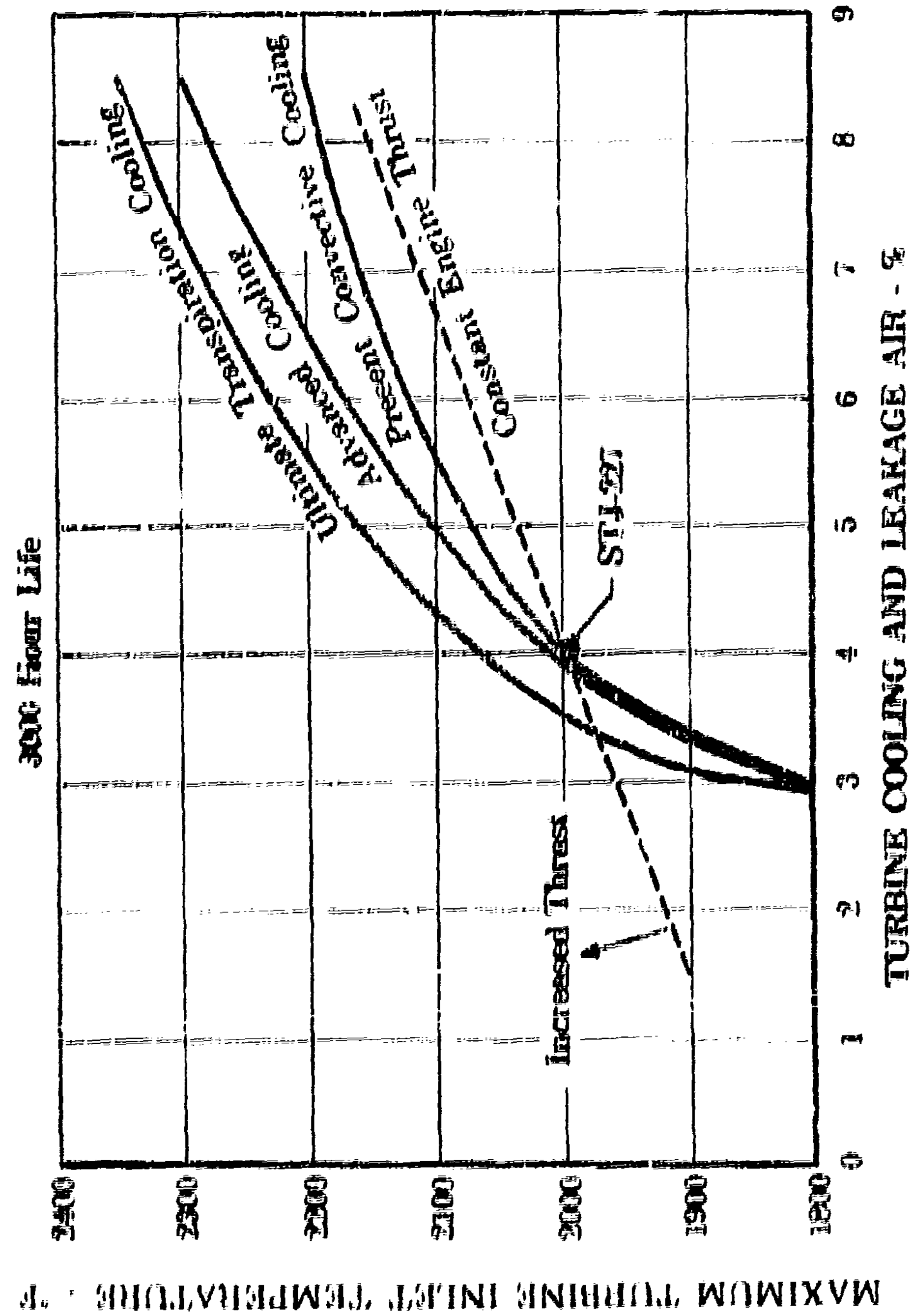

COMBUSTOR EXIT TEMPERATURE PROFILE

Figure 2B-12



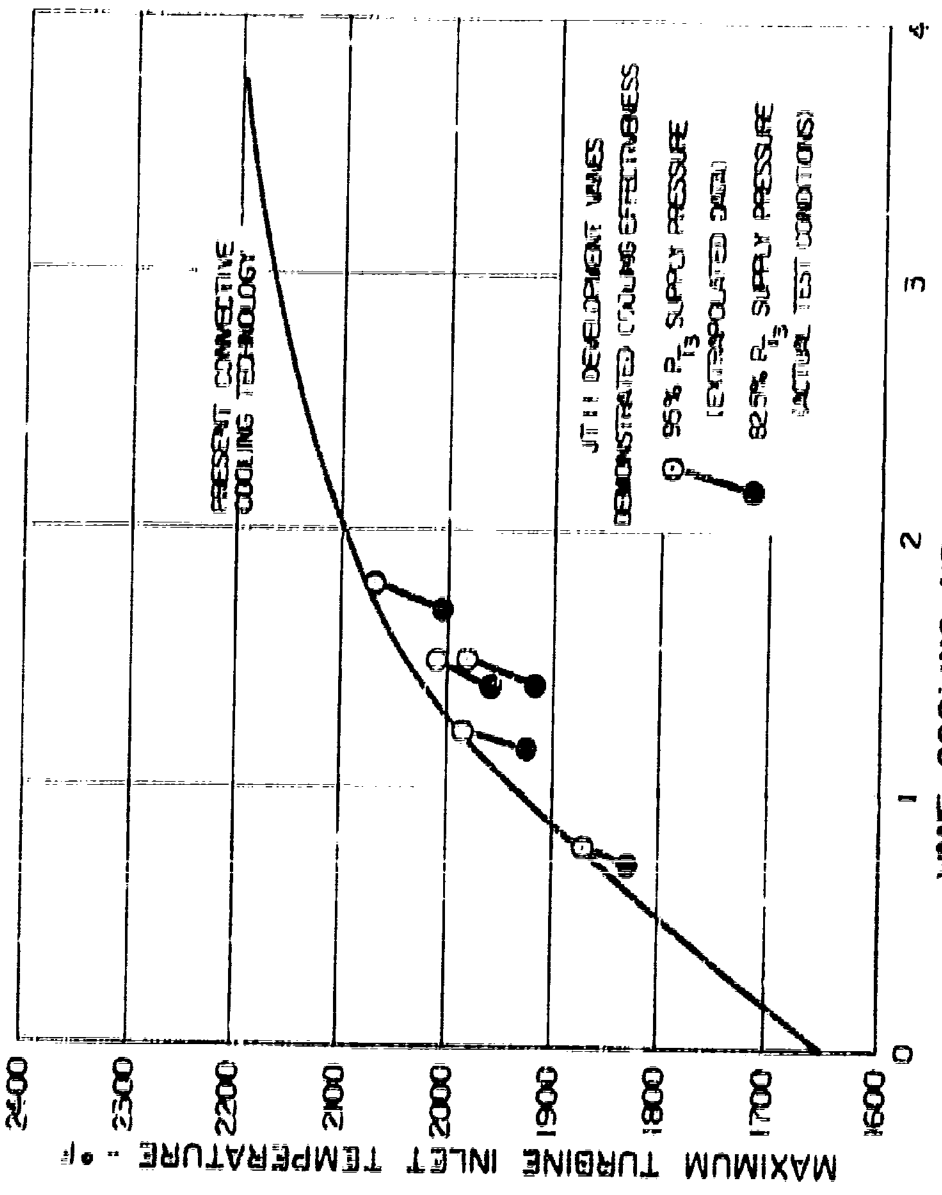

SELECTED COMBUSTOR AIR DISTRIBUTION

Figure 2B-11



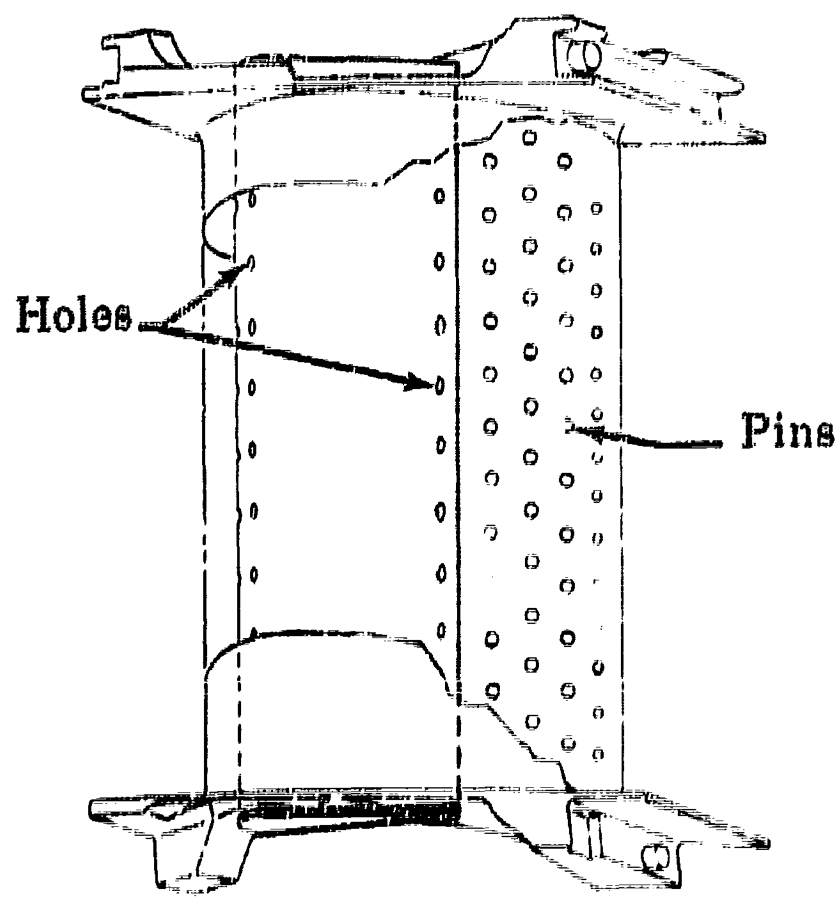

TOTAL TURBINE COOLING AIR REQUIREMENTS

Figure 2B-14



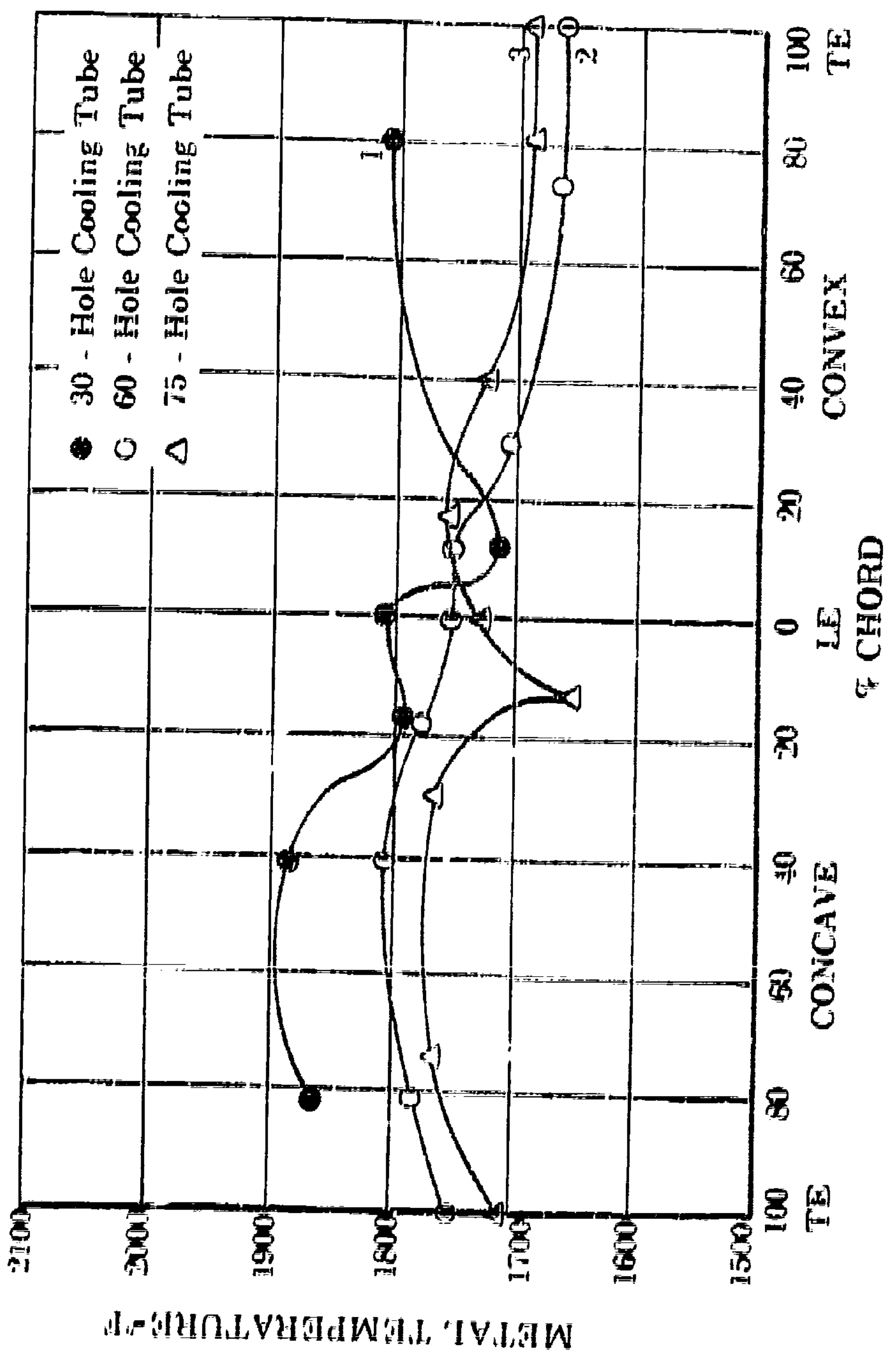

FIRST STAGE TURBINE VANE HEAT BALANCE RESULTS

Figure 2B-15



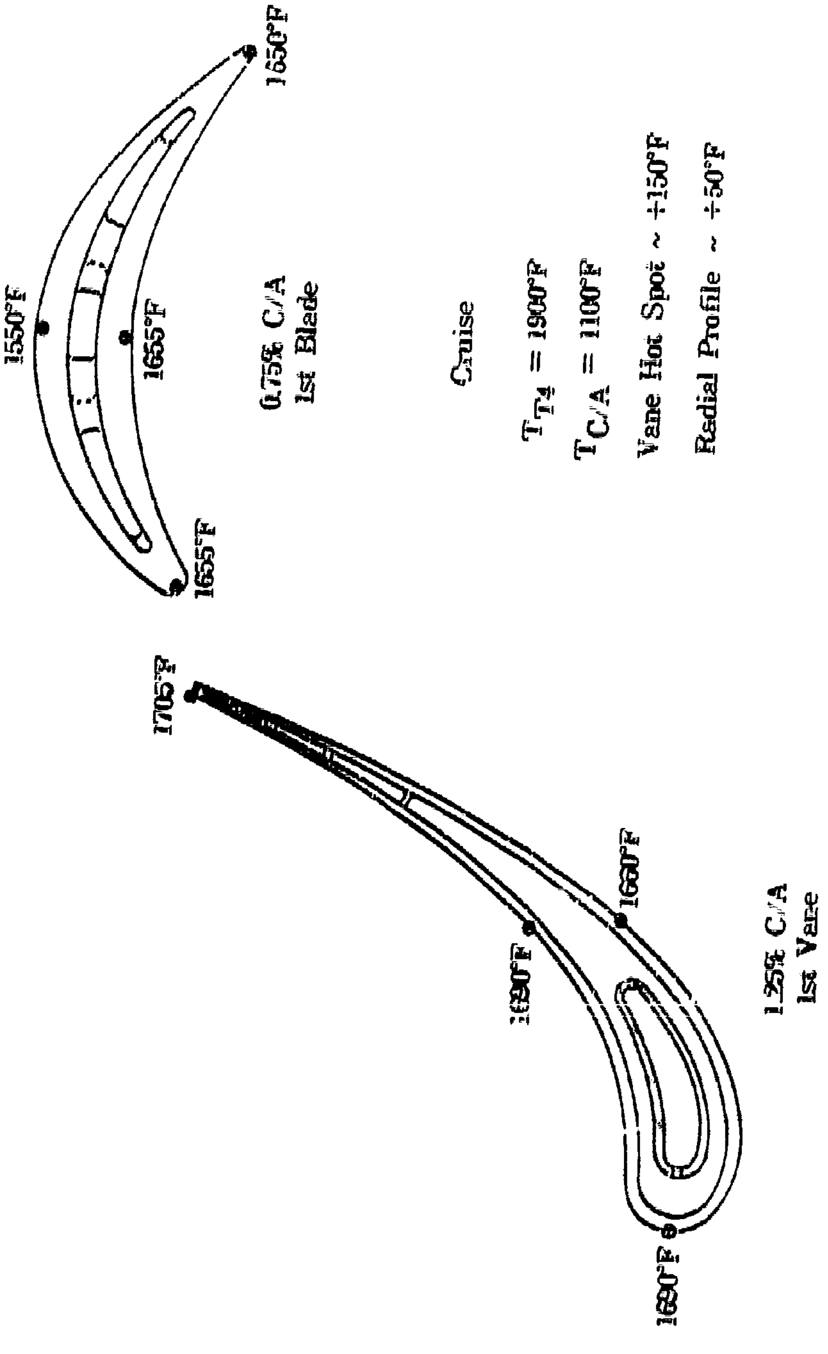

FIRST STAGE TURBINE VANE CONFIGURATION

Figure 2B-16




STATIC HEAT TRANSFER RESULTS

Figure 2B-17




FIRST STAGE ESTIMATED METAL TEMPERATURES

Figure 2B-18

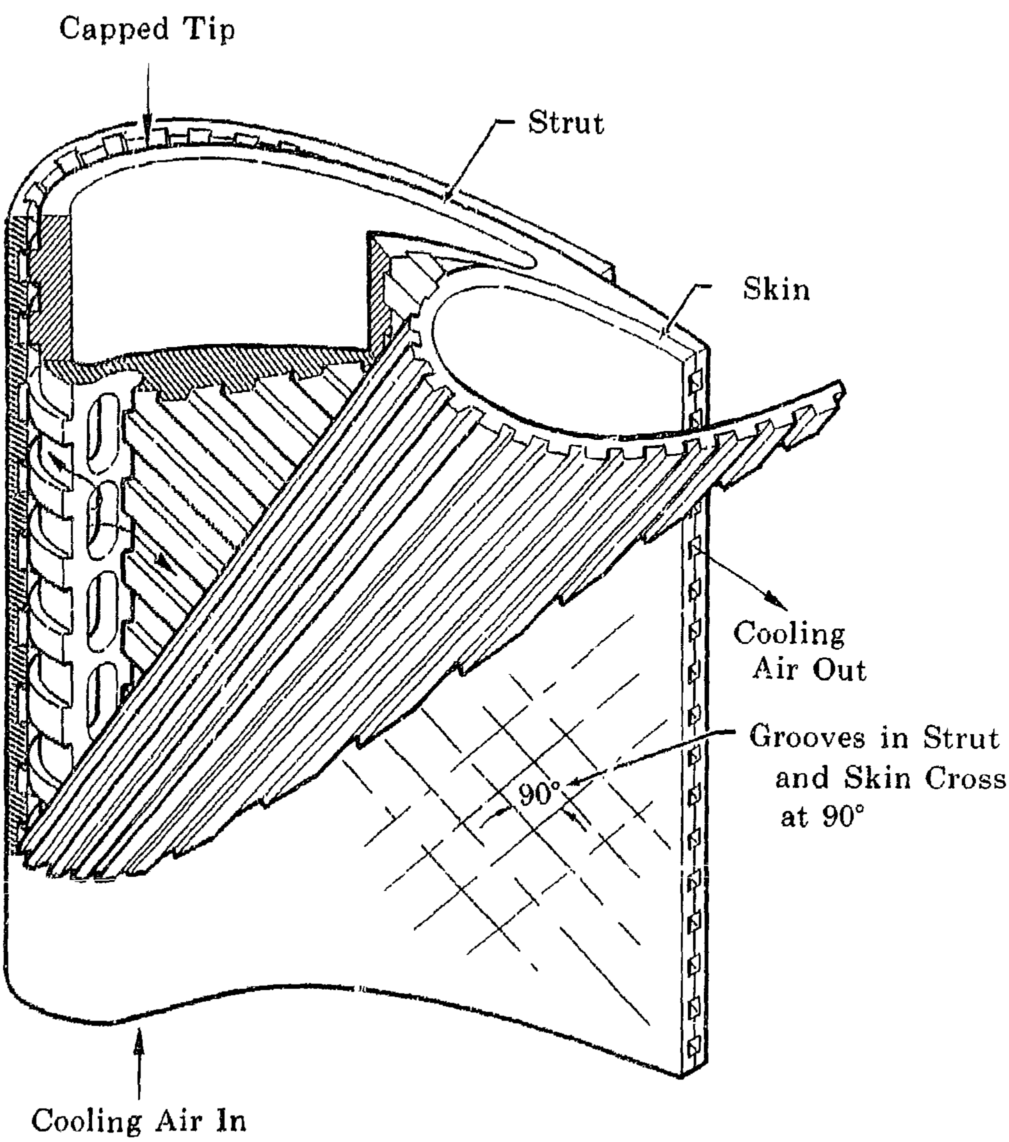
Diamond Baffle

Standard Baffle

Pinned

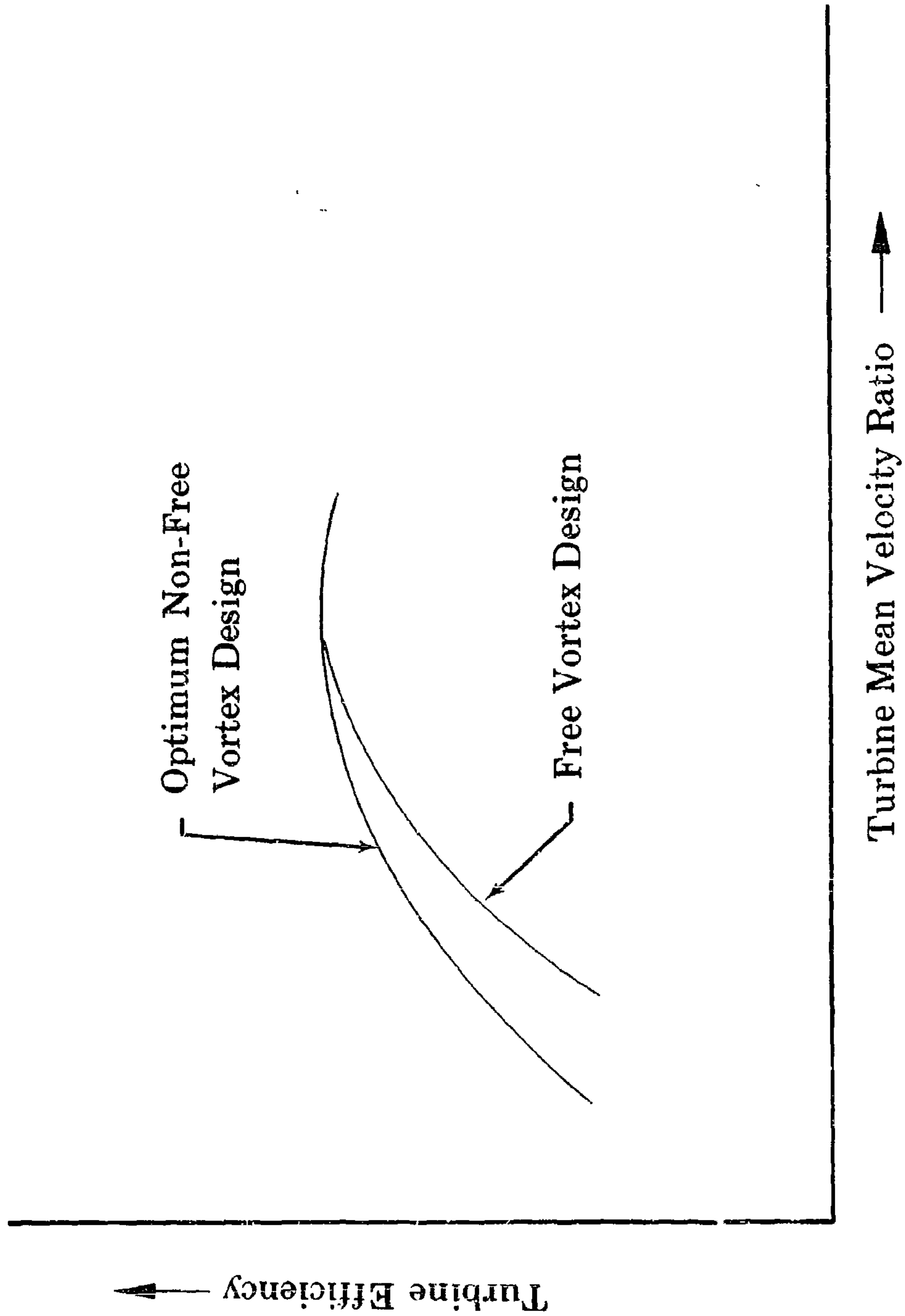
BAFFLE BLADE FLOW CHARACTERISTICS

Figure 2B-19



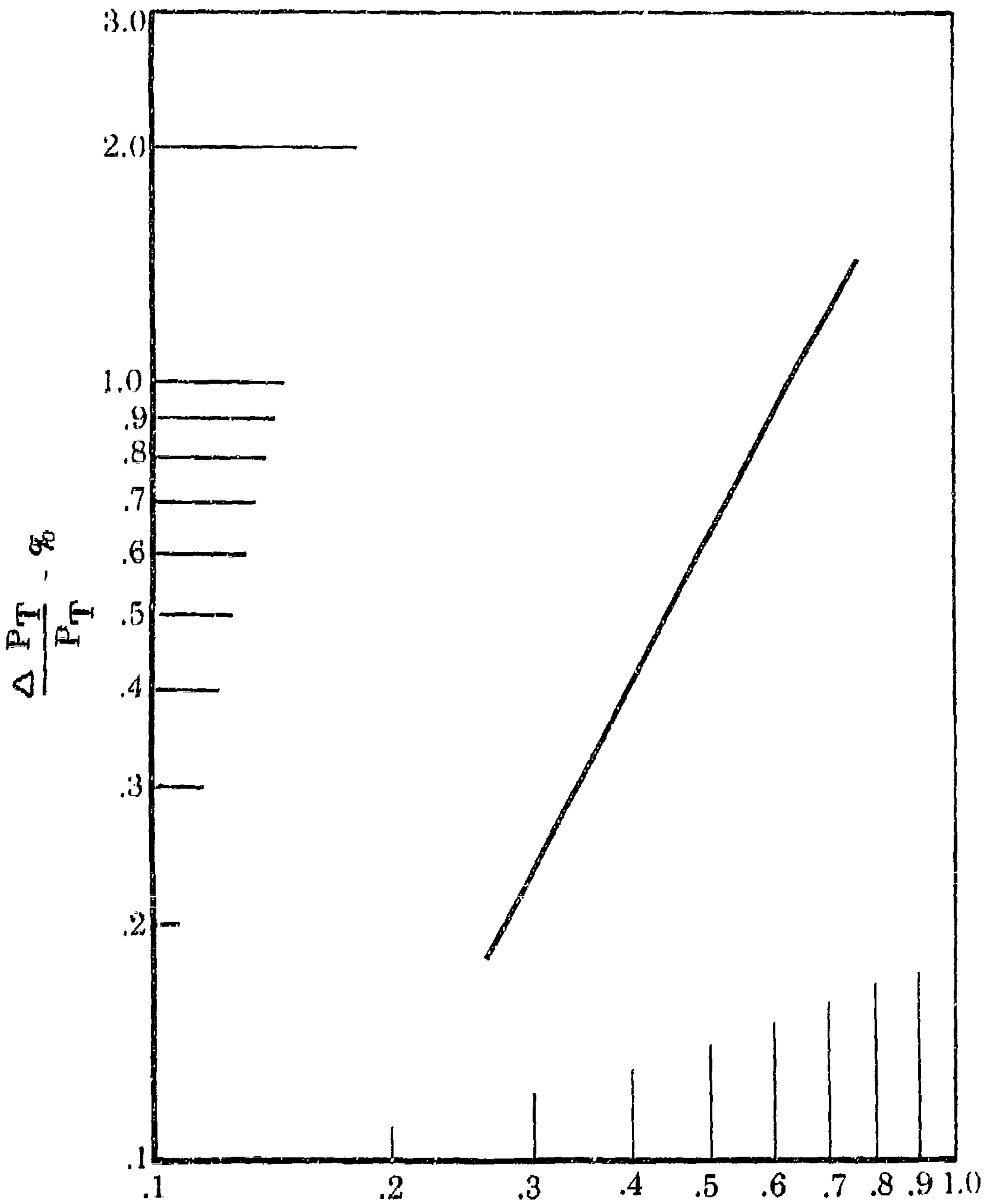

ADVANCED COOLING BLADE SCHEME (FRAENKL SURFACE)

Figure 2B-20



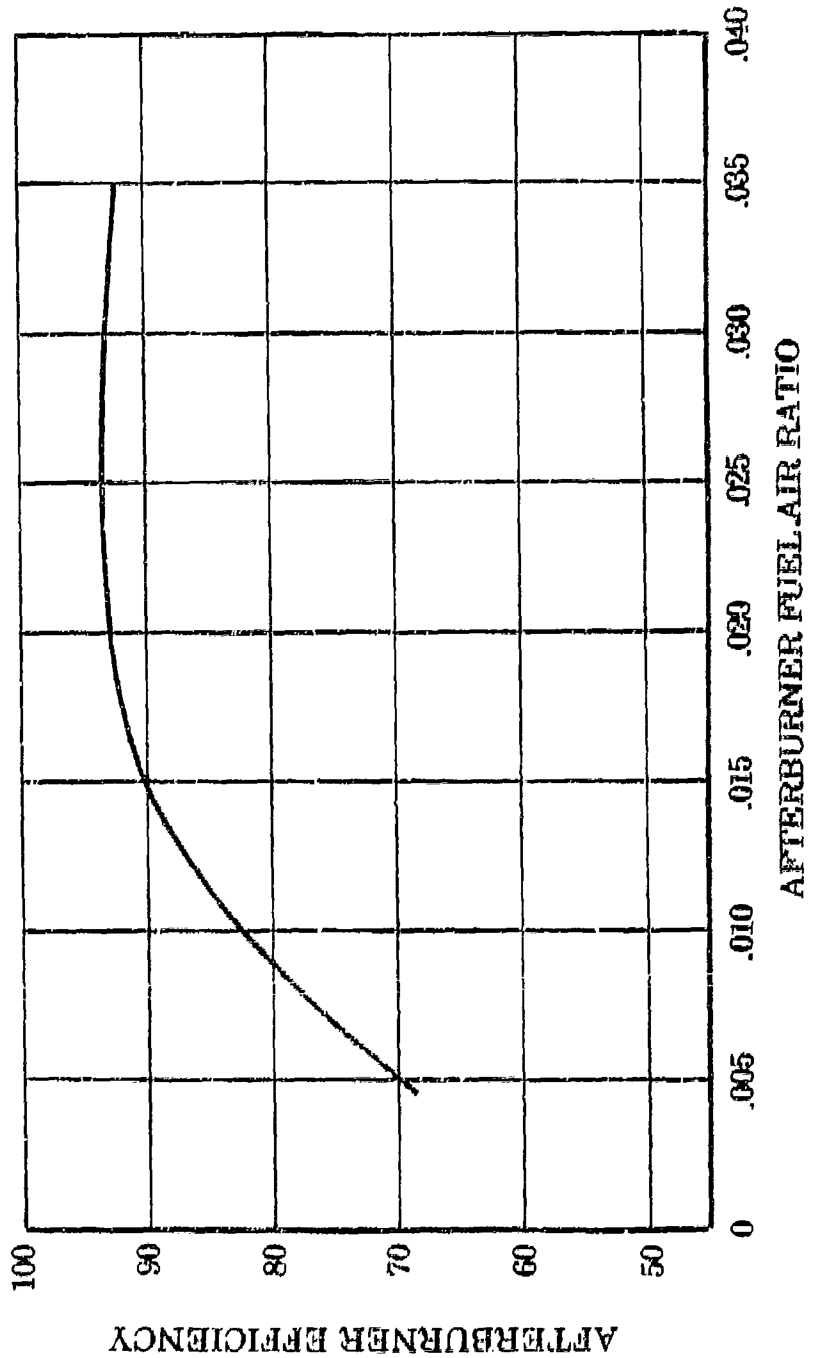

EFFECT OF VORTEX DESIGN ON TURBINE EFFICIENCY

Figure 2B-21



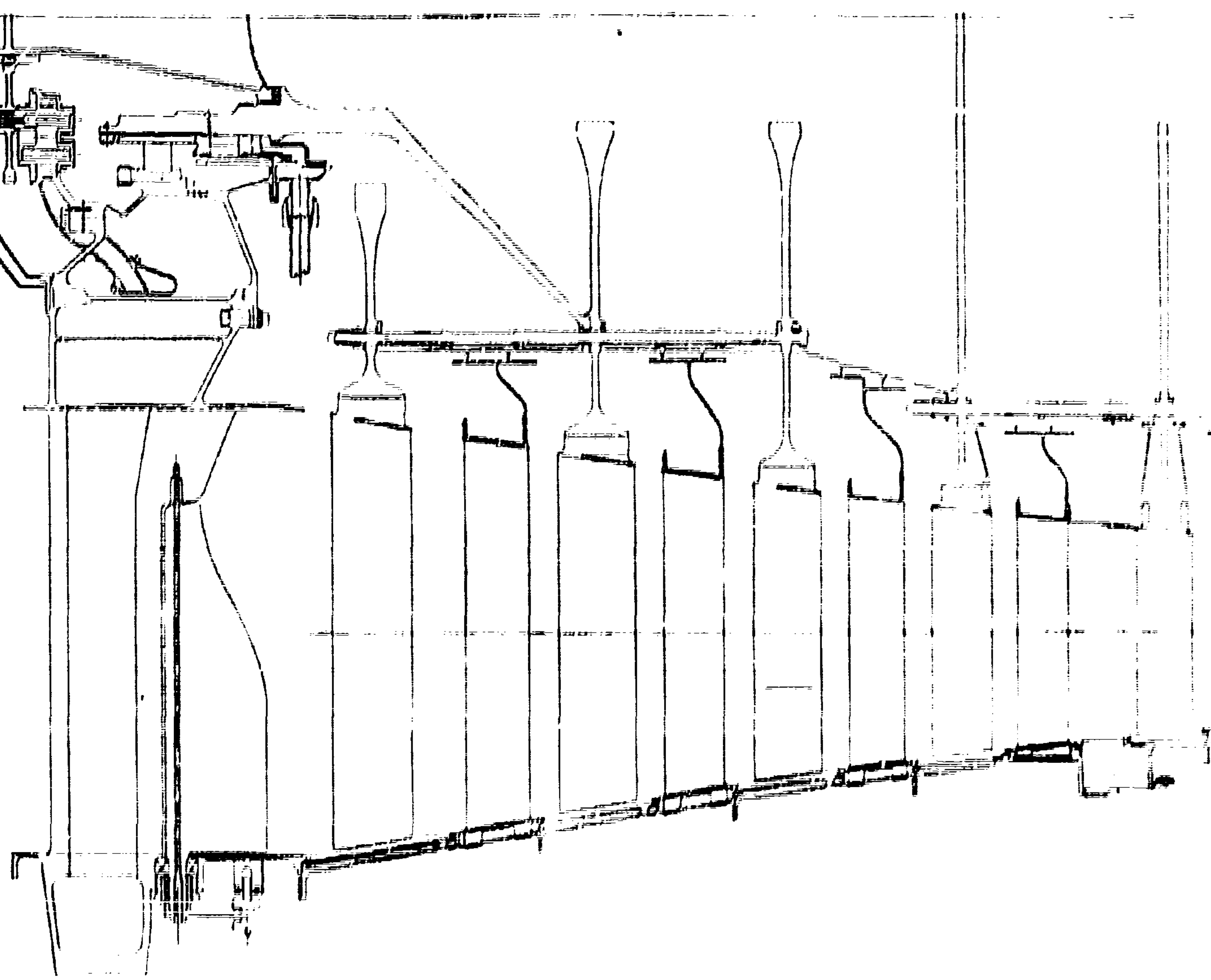

AFTERBURNER COLD FLOW PRESSURE LOSSES

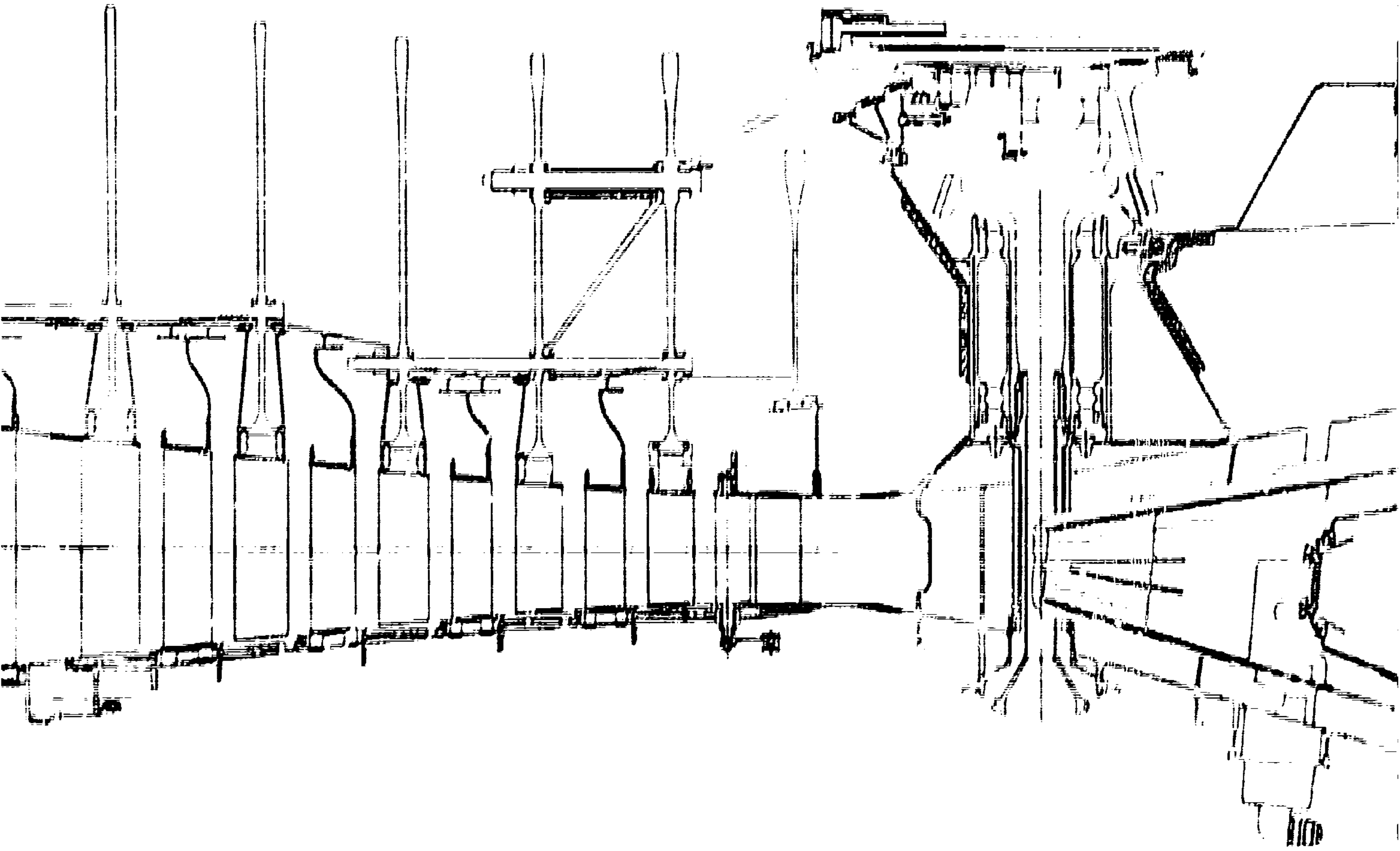
Figure 2B-22

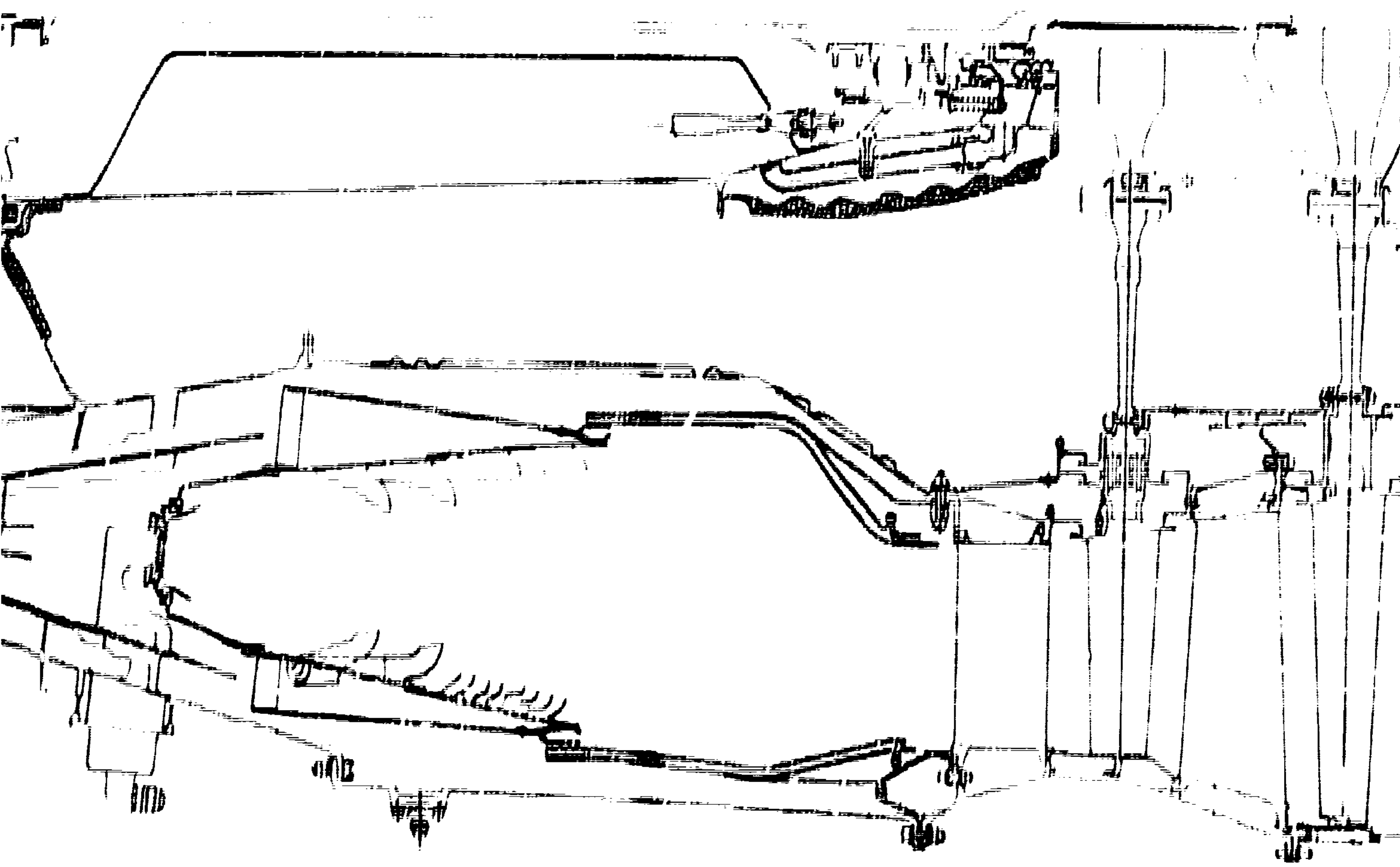



EFFECT OF FUEL/AIR RATIO ON AFTERBURNER EFFICIENCY

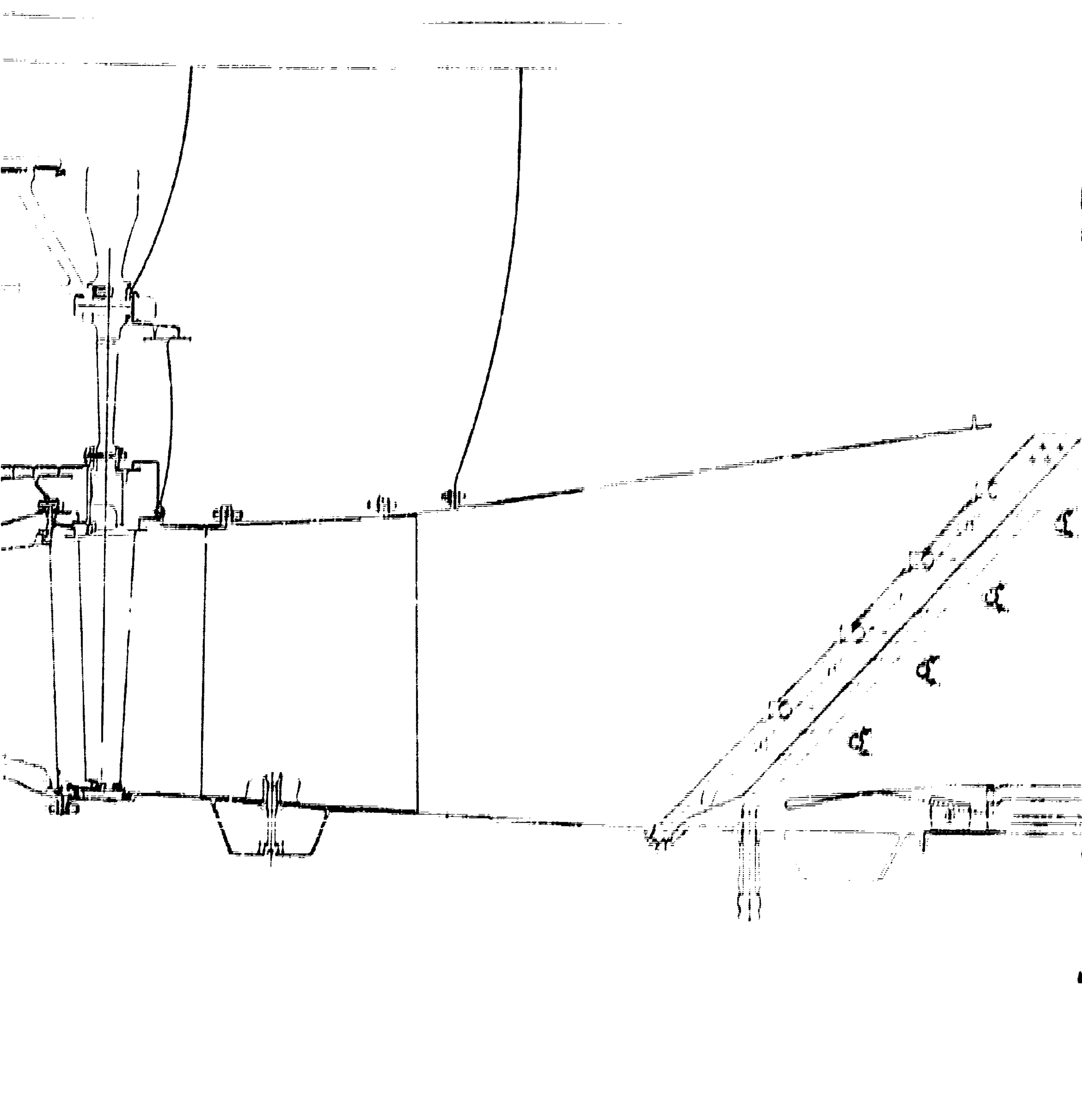
Figure 2B-23

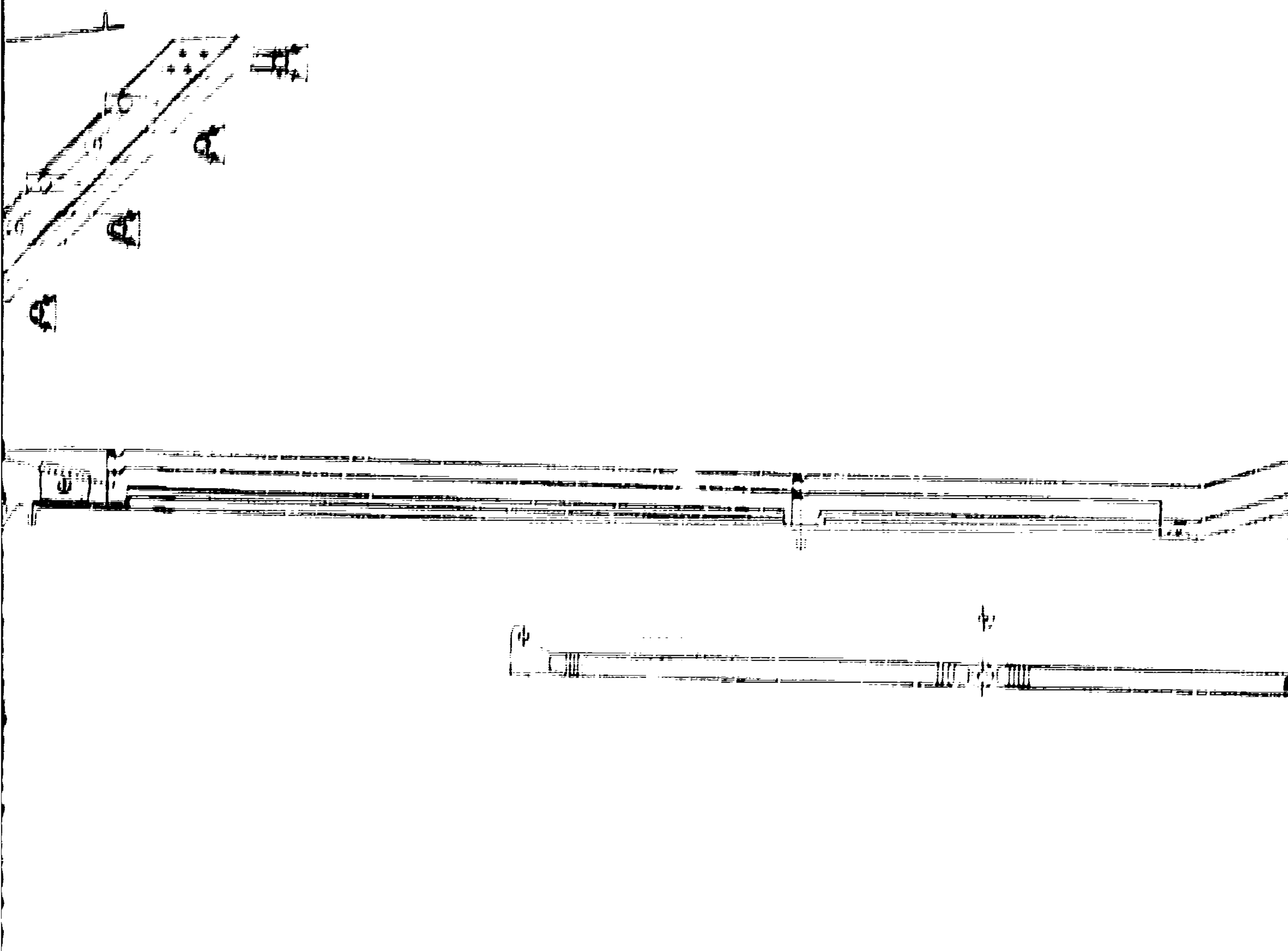


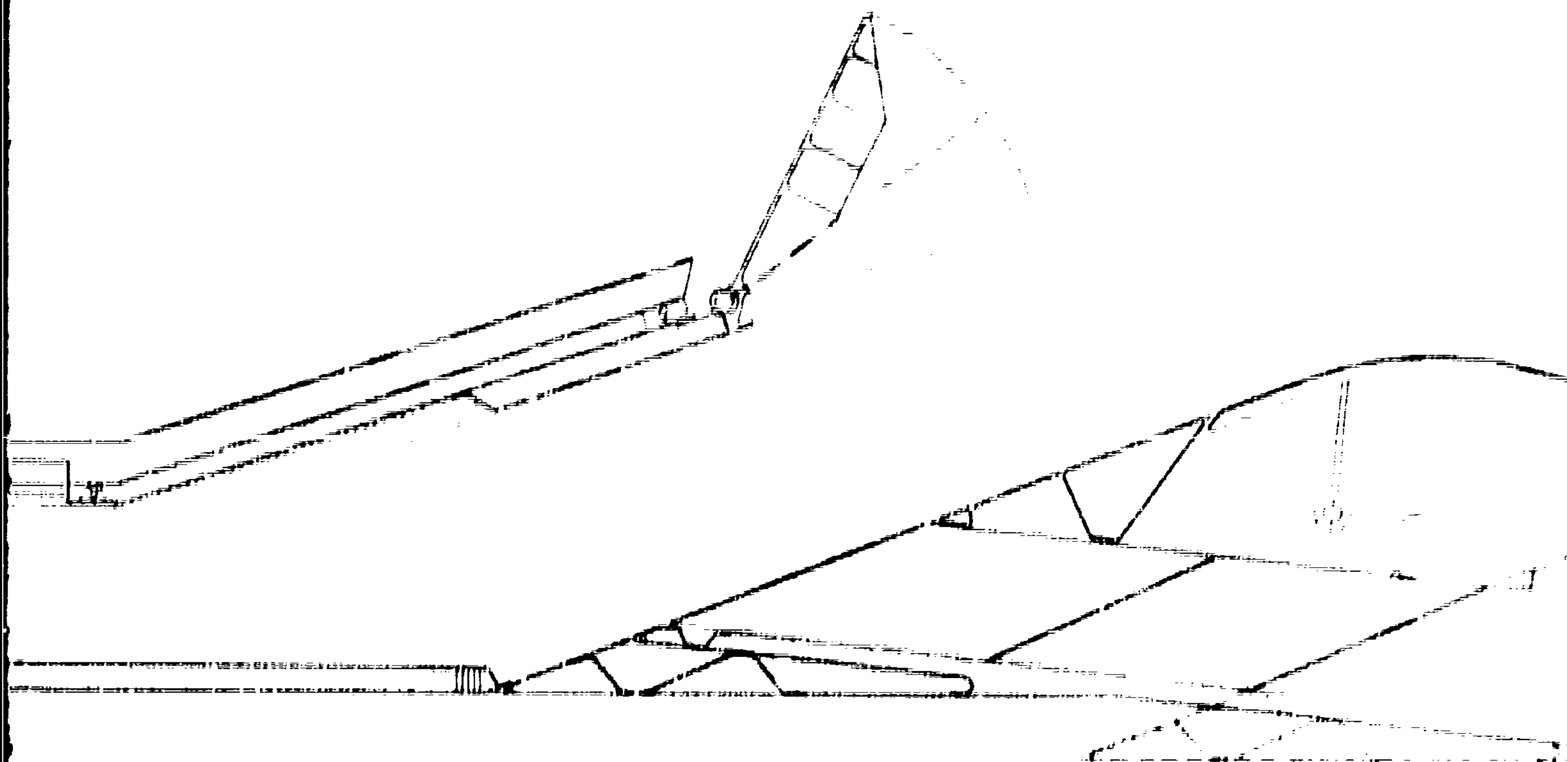


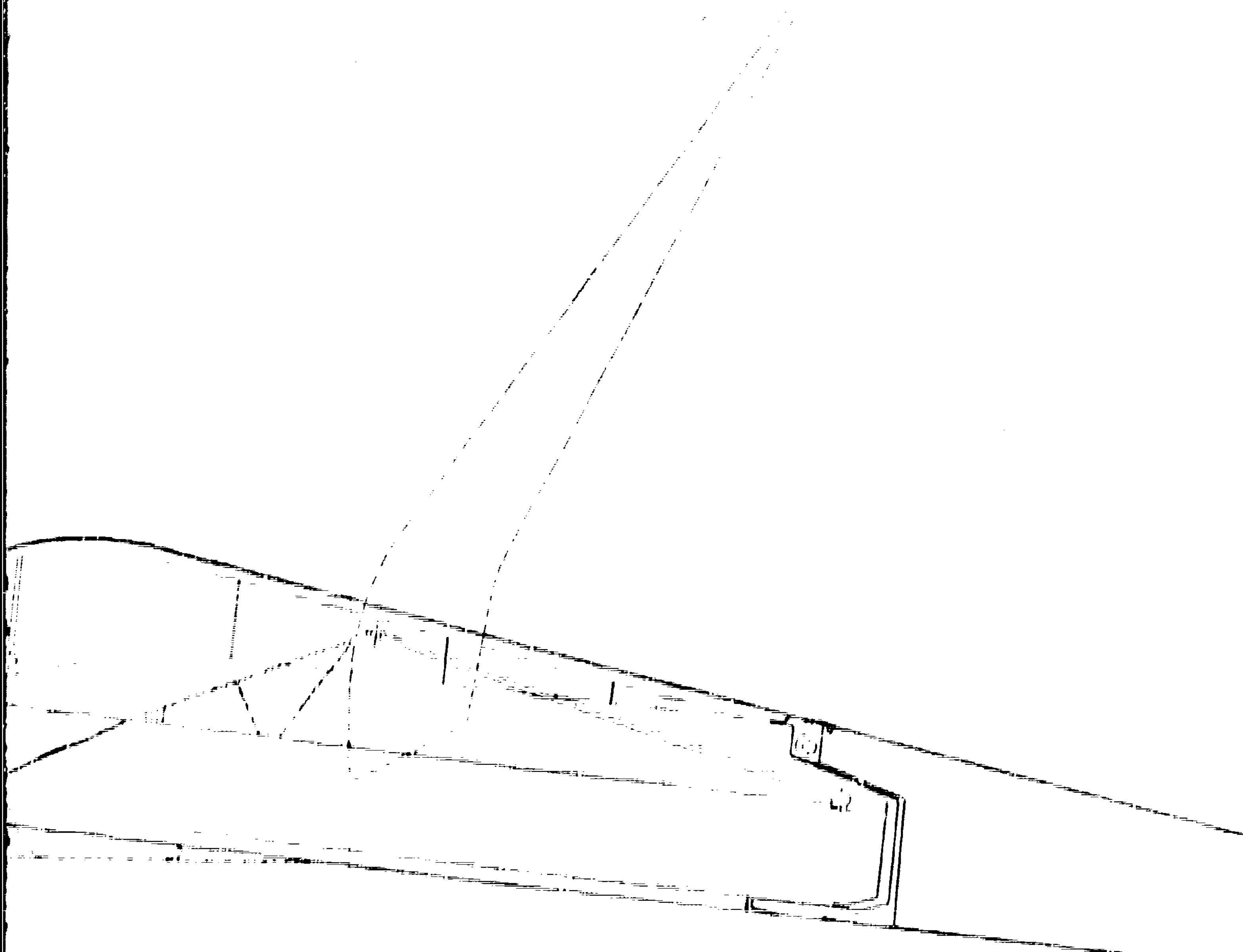


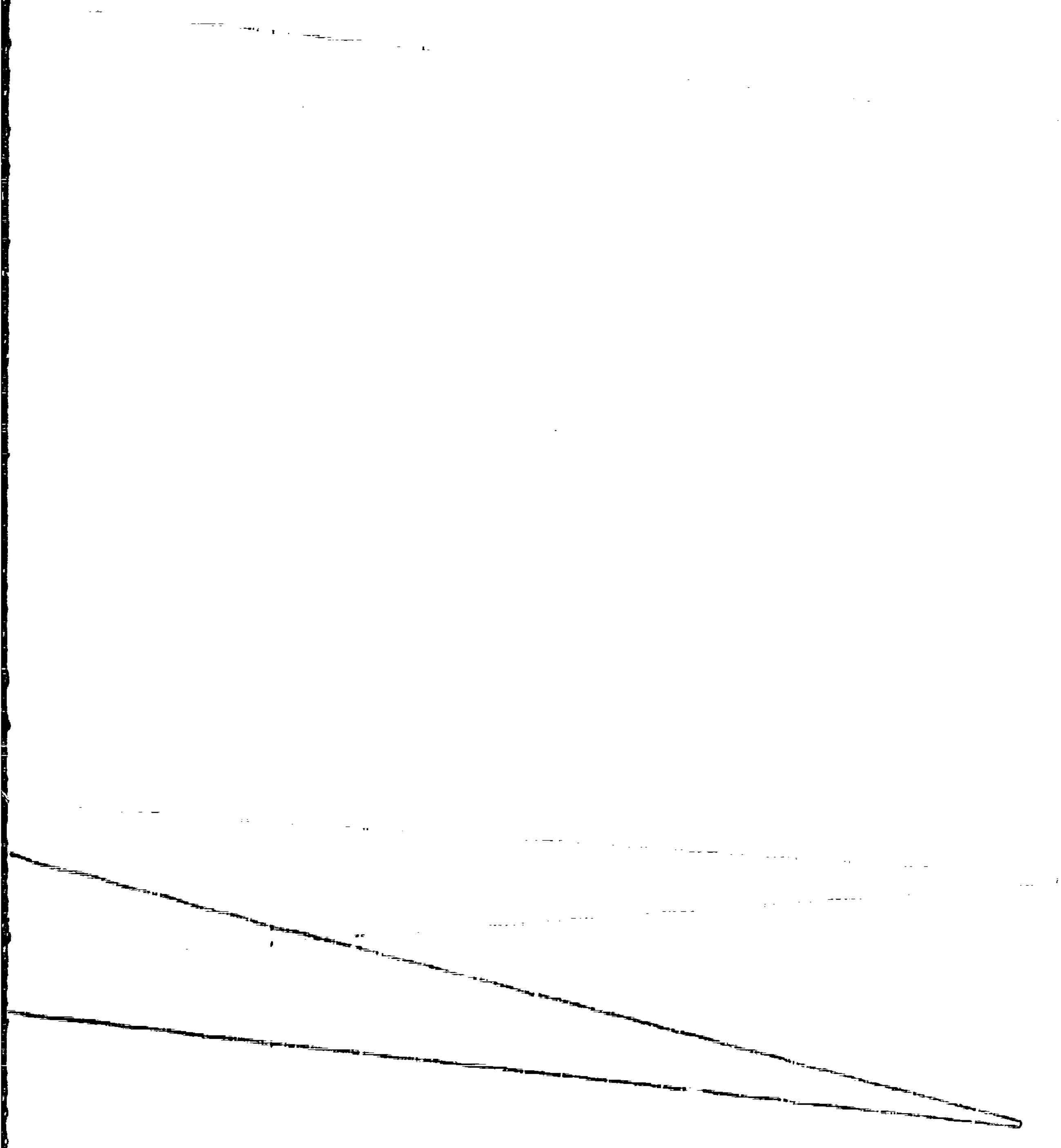
3





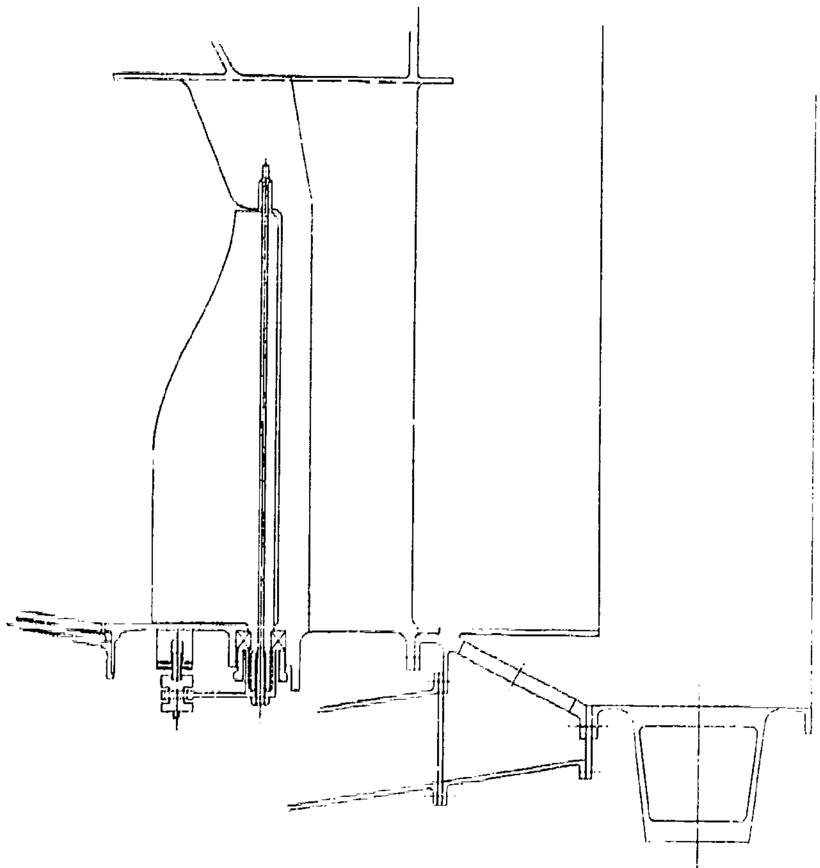






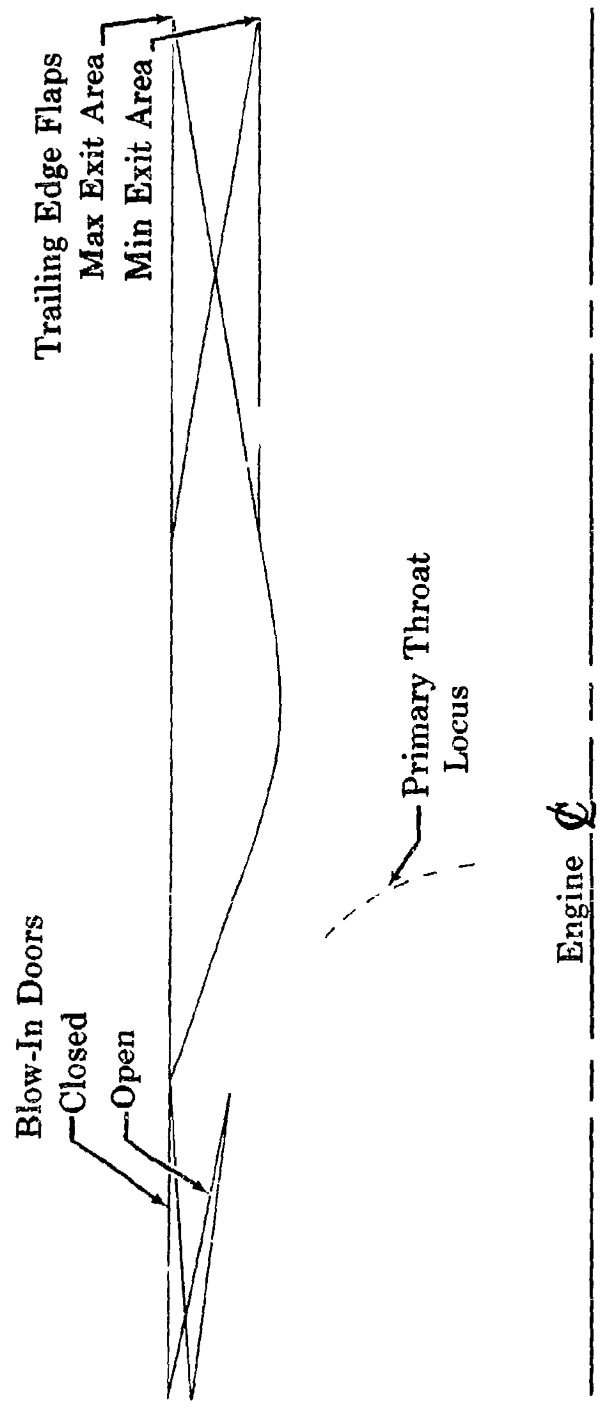
STJ227 CROSS-SECTION DRAWING

Figure 2B-24



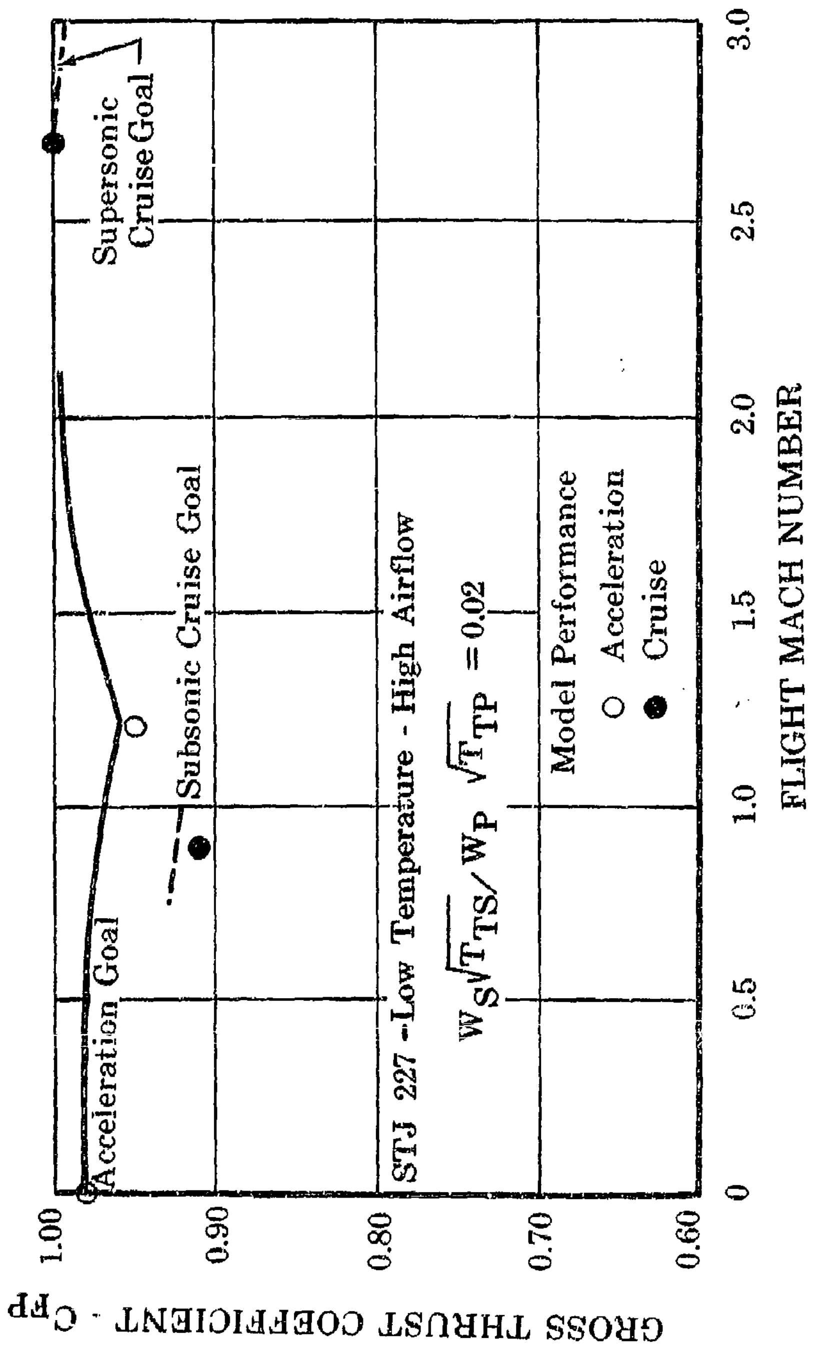
STJ227 CROSS-SECTION DRAWING

Figure 2B-24 (Cont'd)



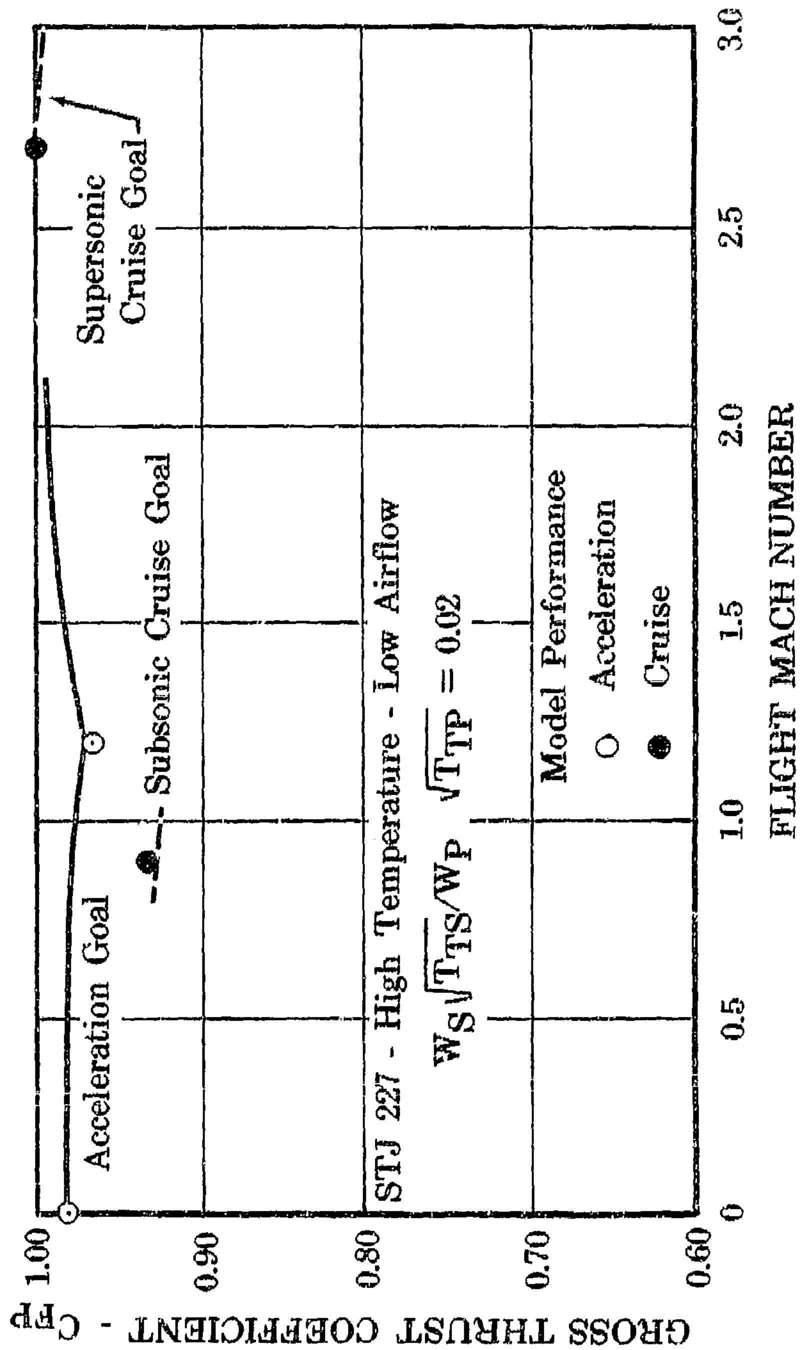

TYPICAL FULL AFTERBURNING TURBOJET EJECTOR

Figure 2B-25



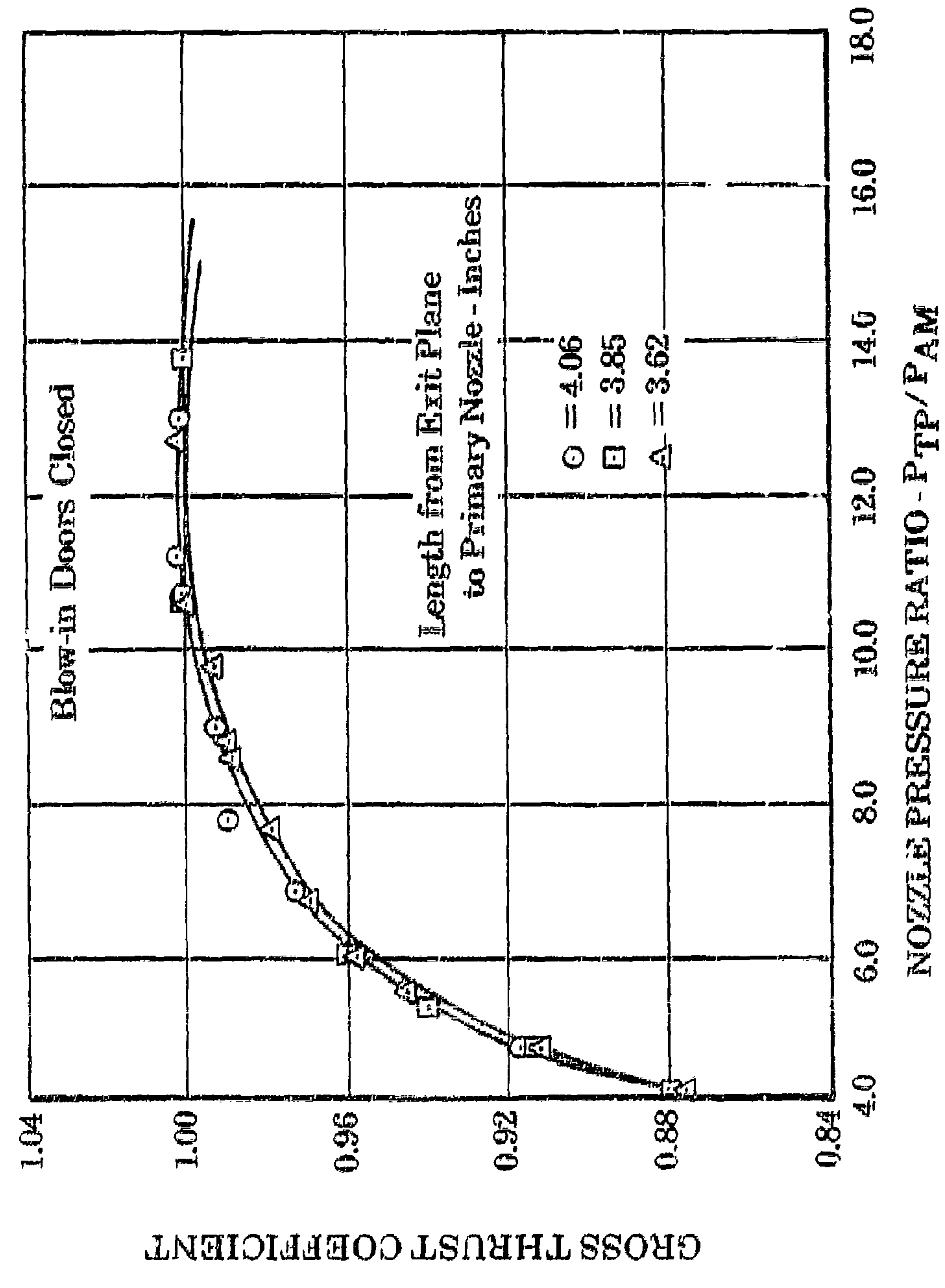

COMPARISON OF SCALE MODEL BLOW-IN DOOR
EJECTOR WITH GOAL VALUES

Figure 2B-26



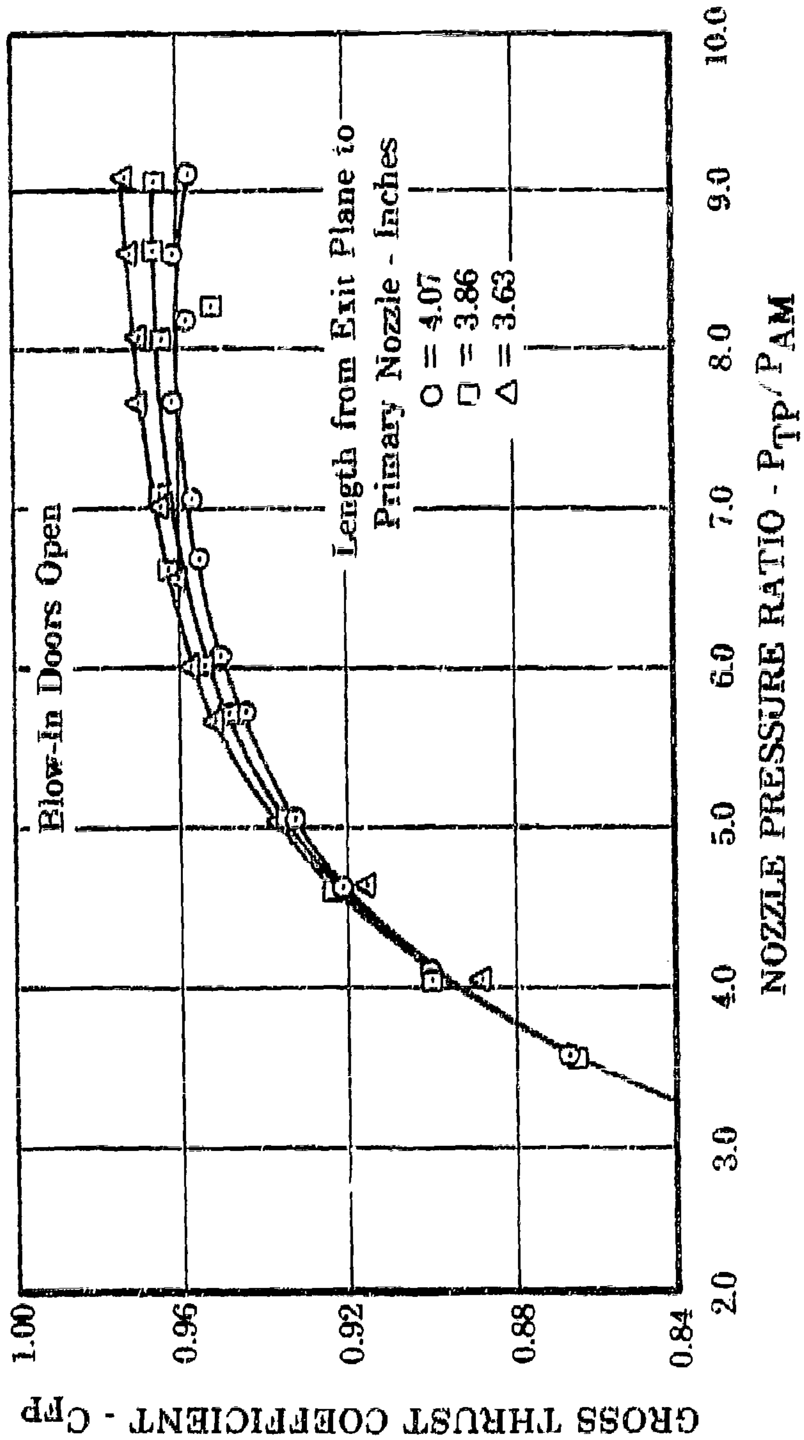

COMPARISON OF SCALE MODEL BLOW-IN DOOR EJECTOR WITH GOAL VALUES

Figure 2B-27



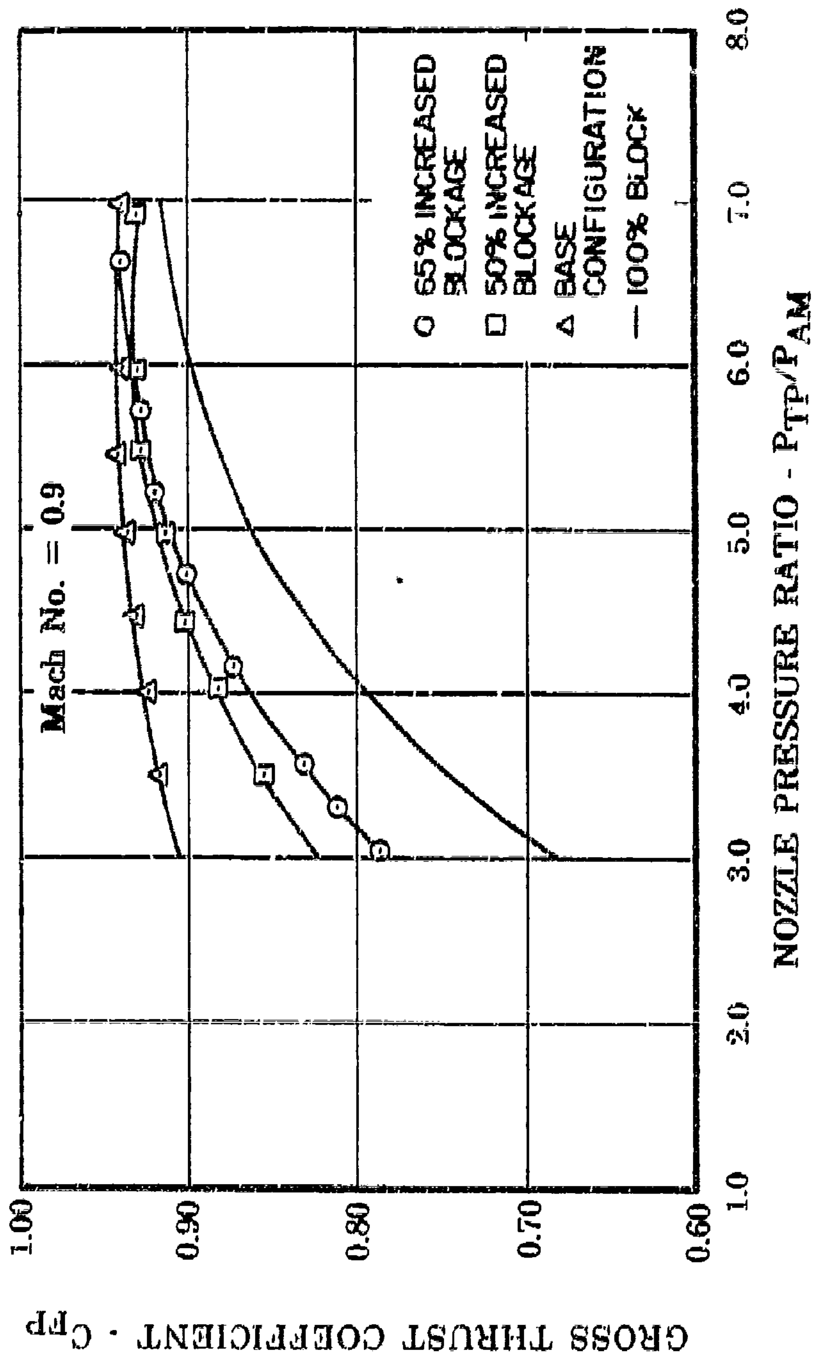

EFFECT OF PRIMARY NOZZLE SPACING
ON EJECTOR PERFORMANCE

Figure 2B-28



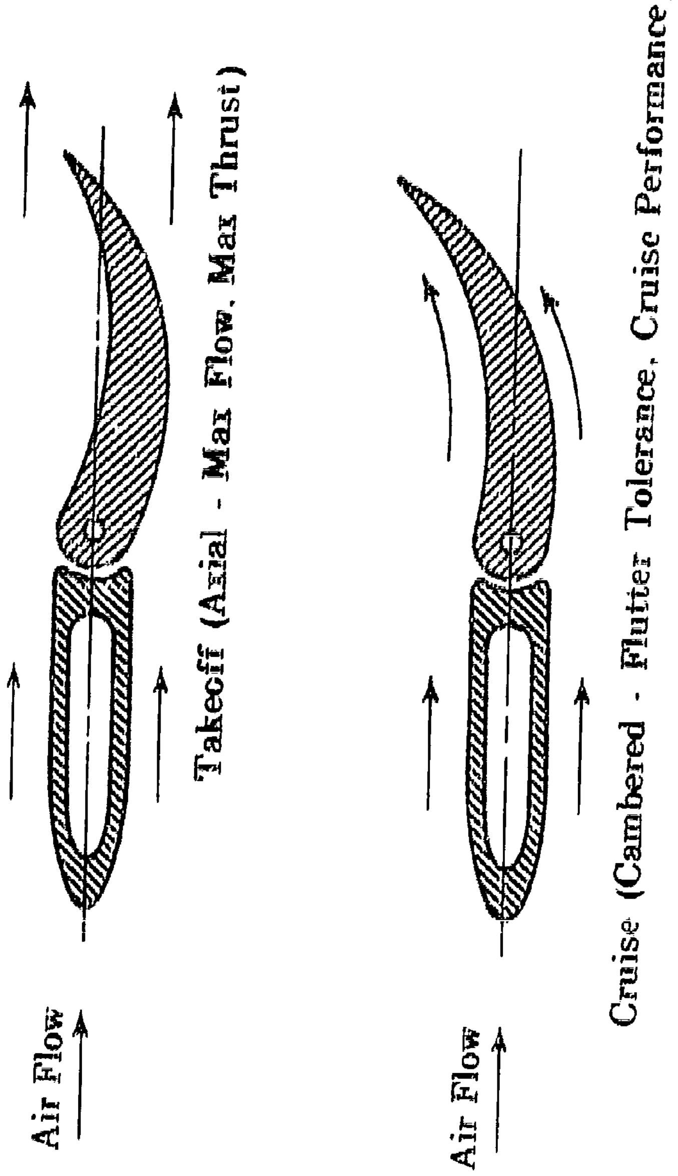

EFFECT OF PRIMARY NOZZLE SPACING
ON EJECTOR PERFORMANCE

Figure 2B-29



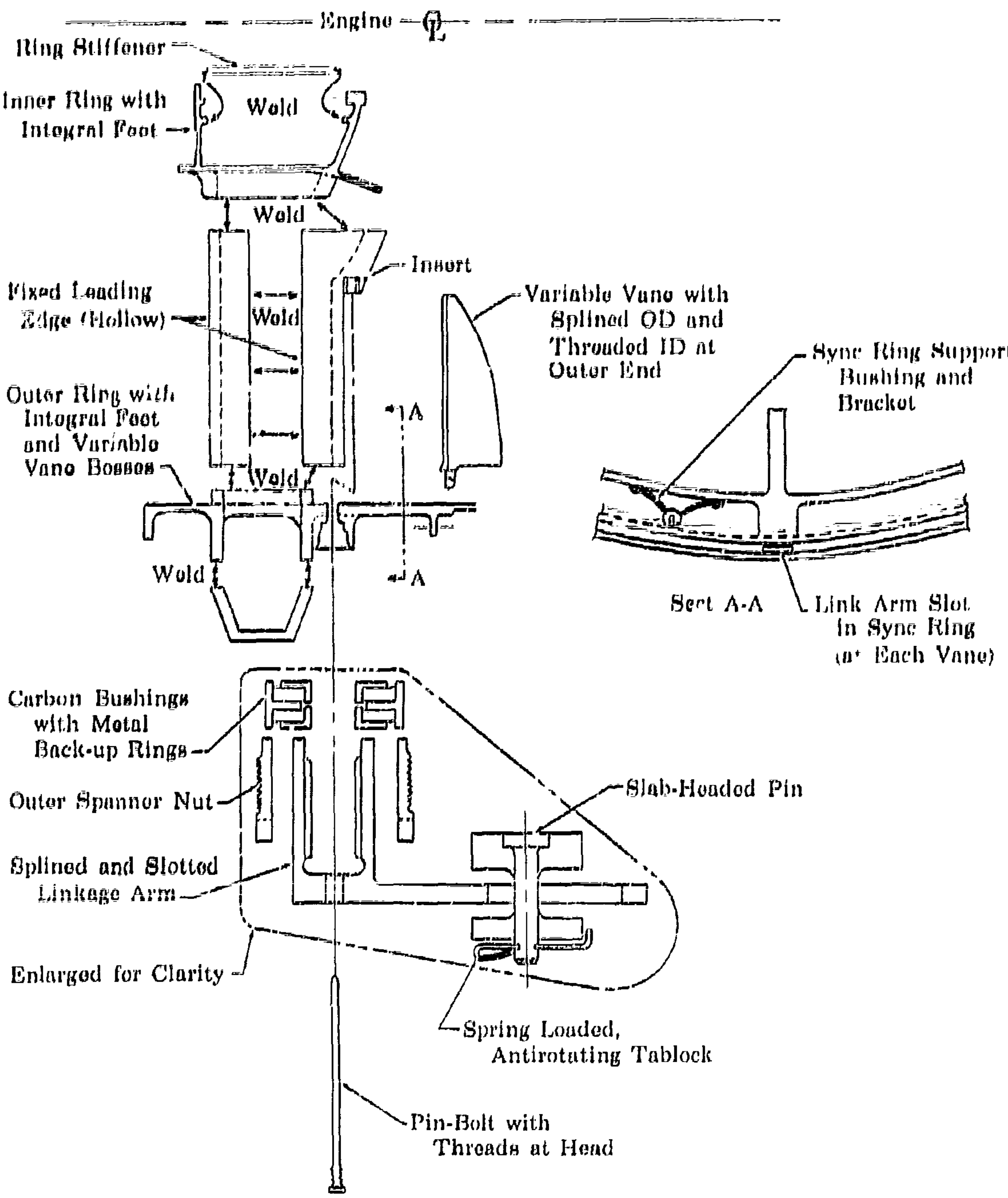

EFFECT OF REDUCED TERTIARY AIRFLOW

Figure 2B-30



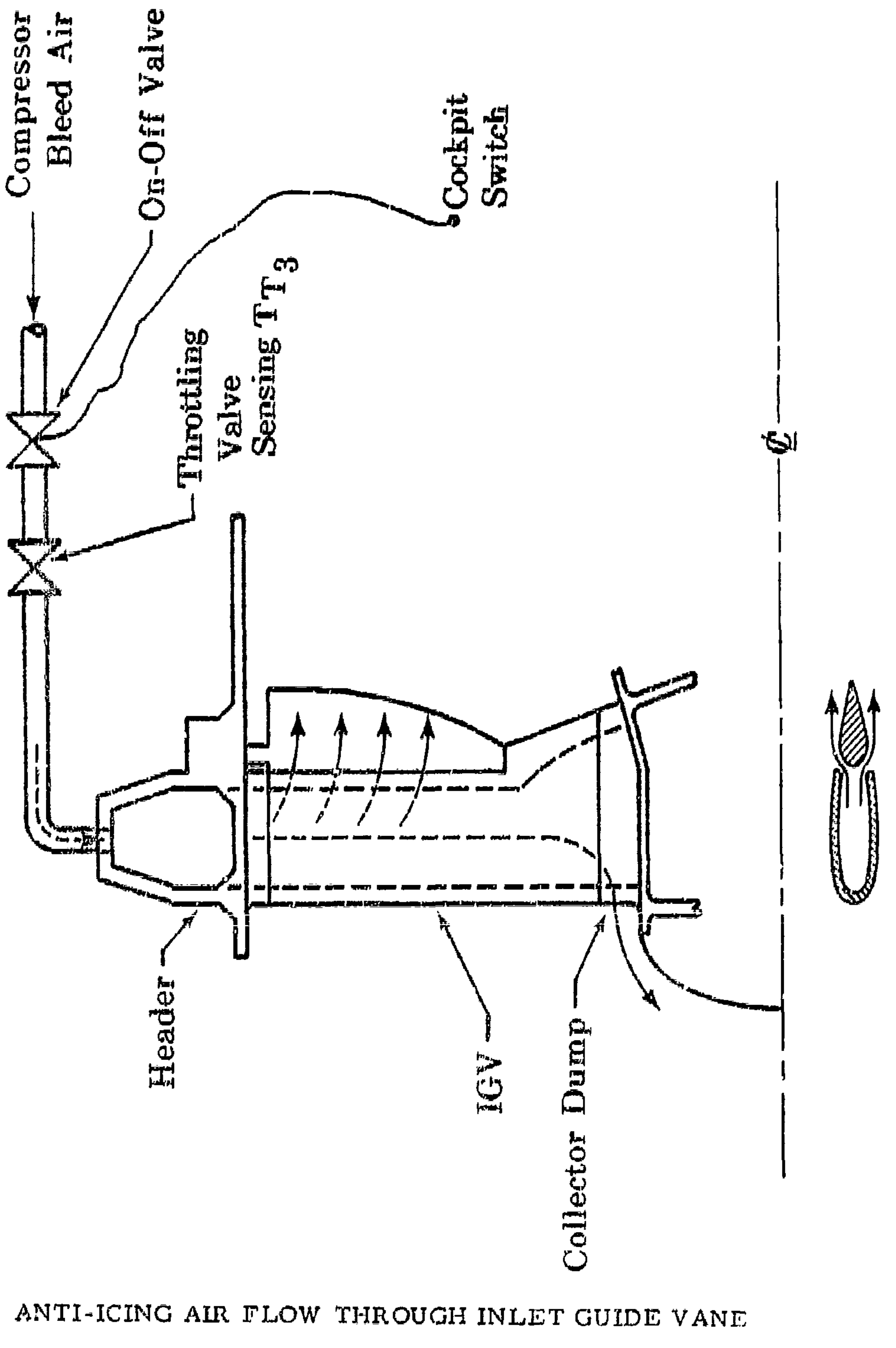

VARIABLE INLET GUIDE DIAGRAM

Figure 2B-31



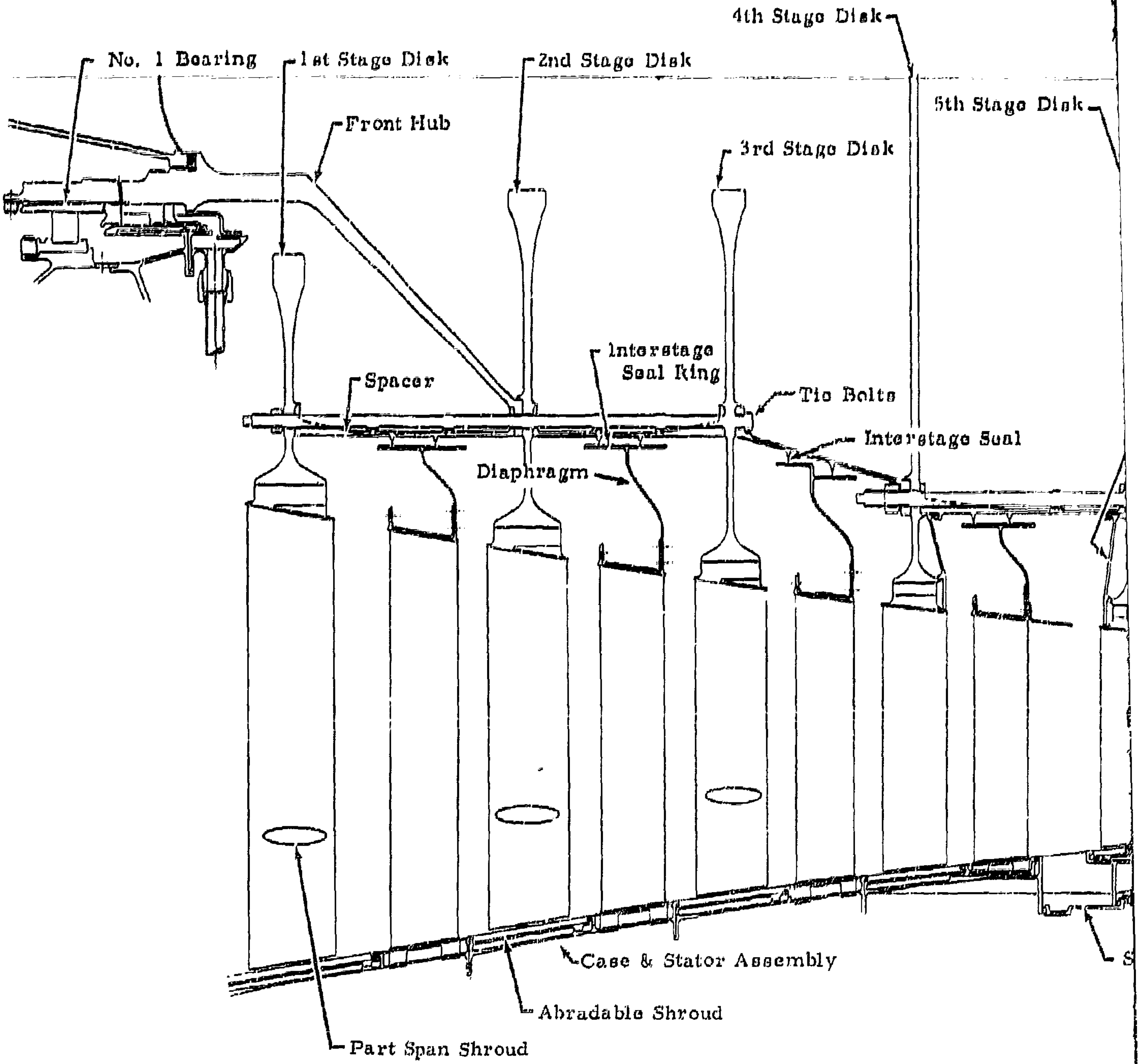

GENERAL INLET CASE CONSTRUCTION AND ASSEMBLY

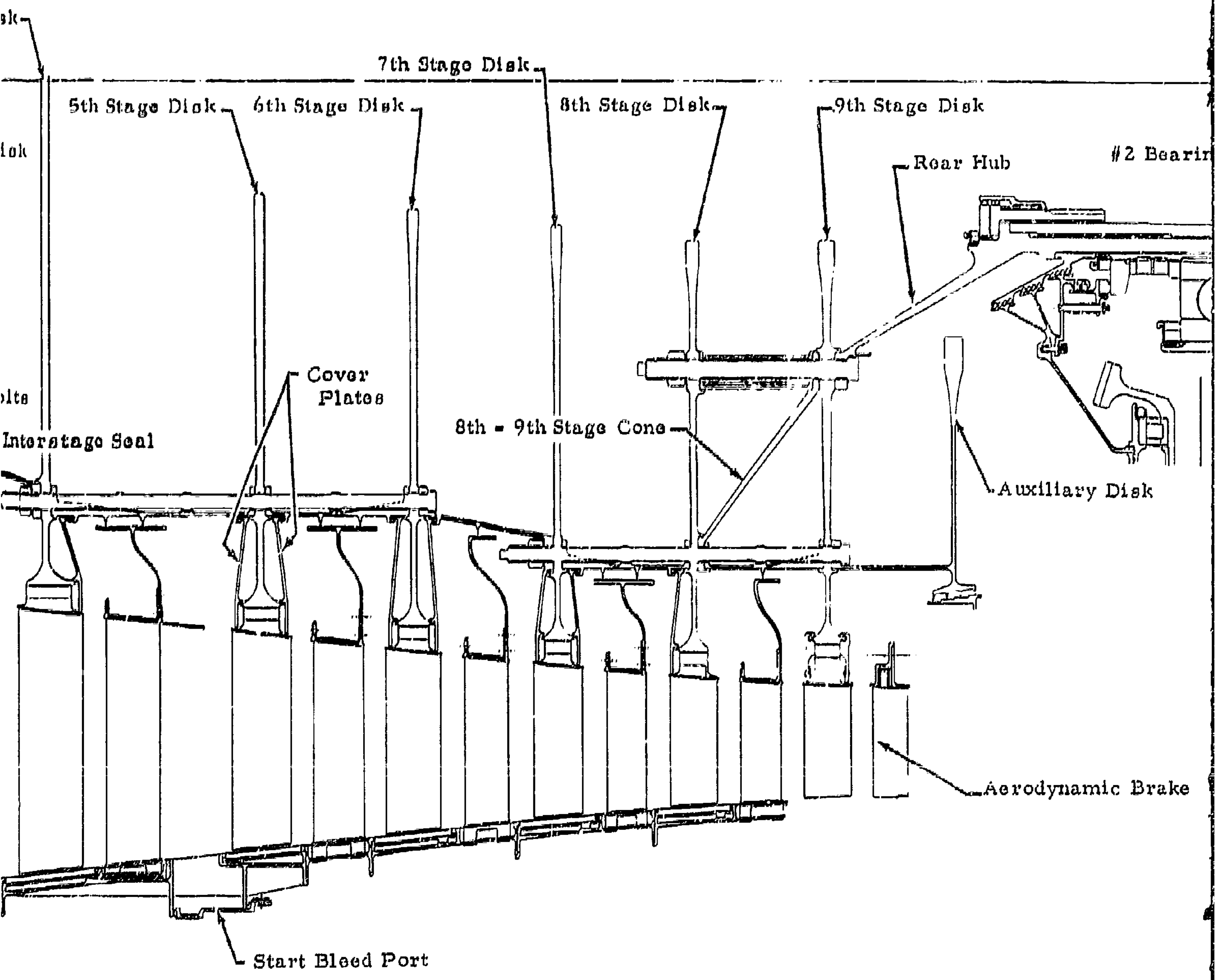
Figure 2B-32



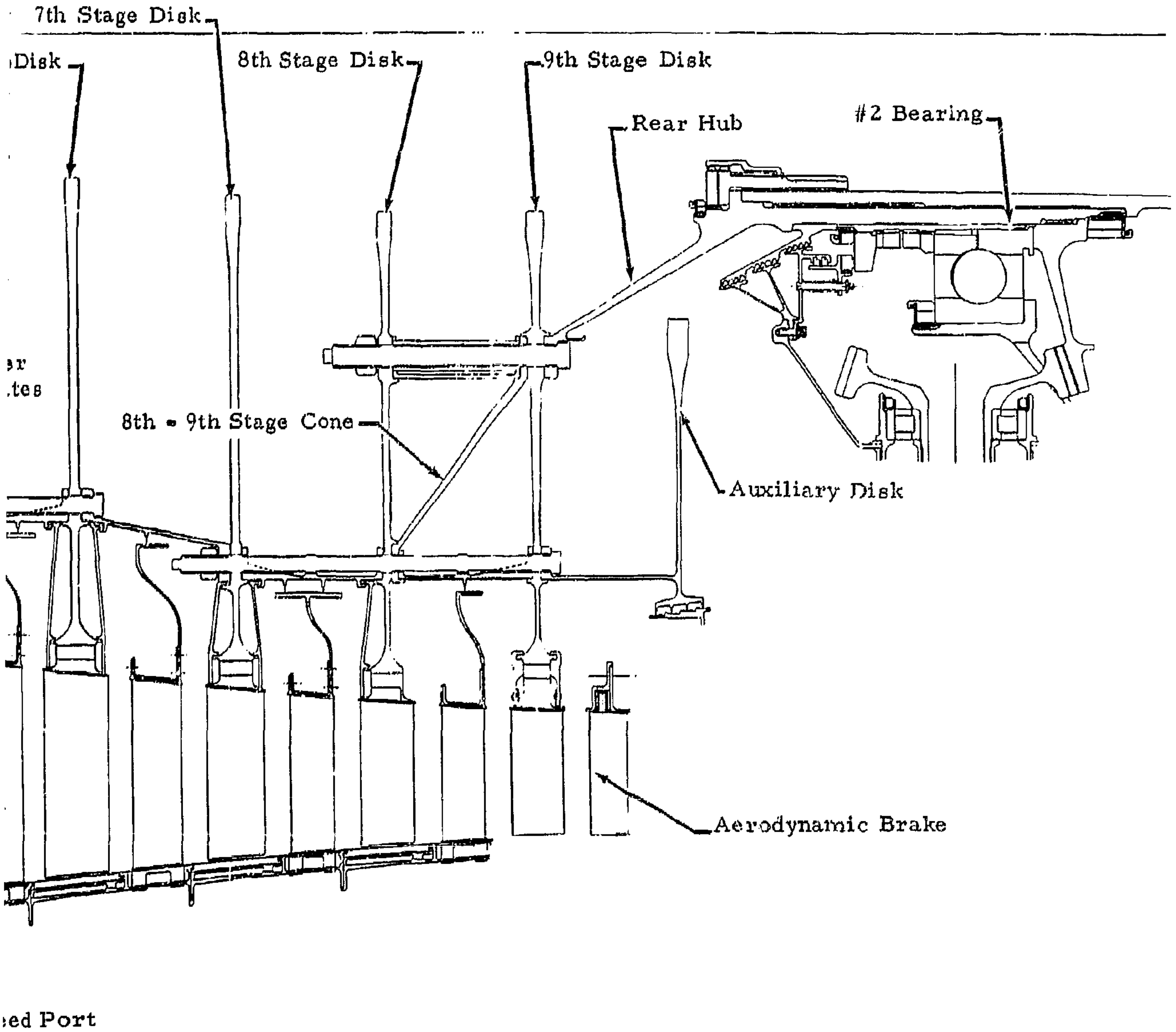

ANTI-ICING AIR FLOW THROUGH INLET GUIDE VANE

Figure 2B-33

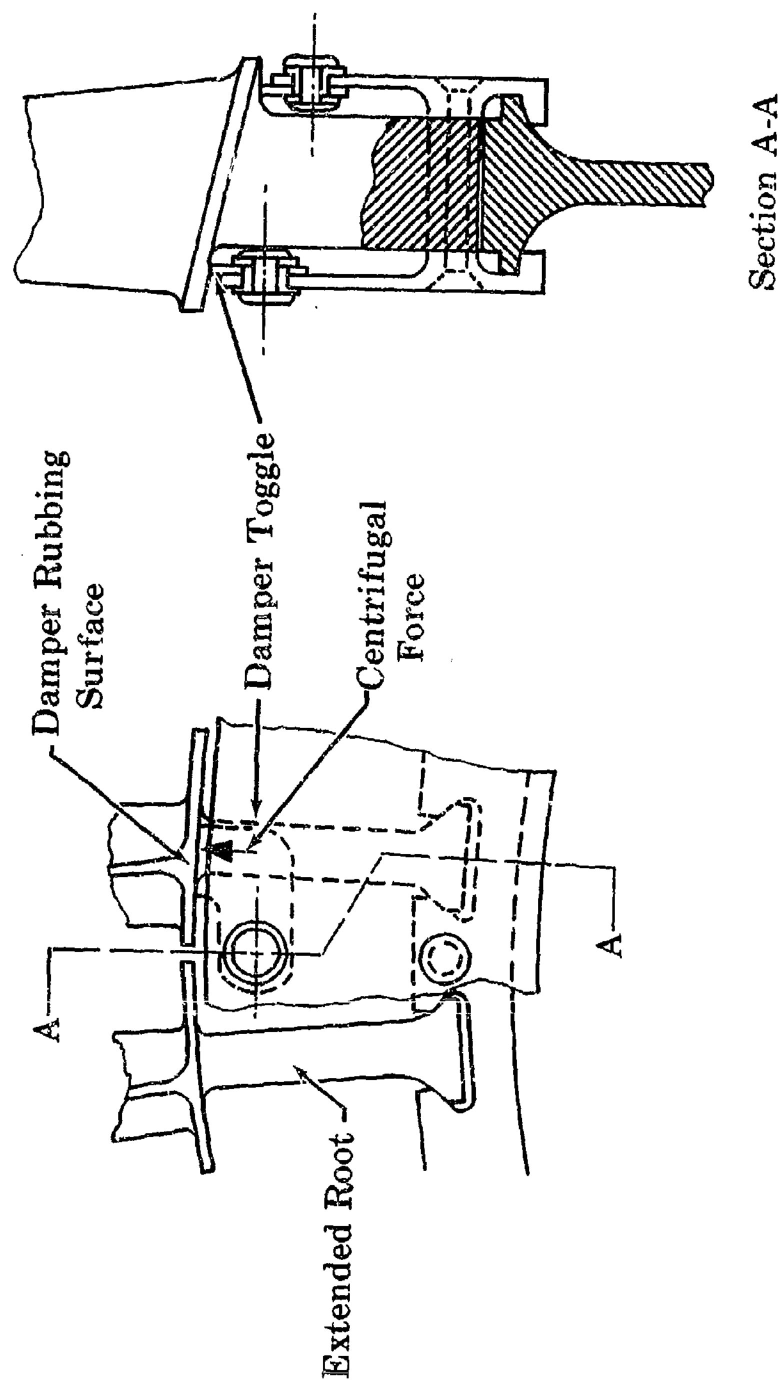

4th Stage Disk
No. 1 Bearing
1st Stage Disk
2nd Stage Disk
5th Stage Disk
Front Hub
3rd Stage Disk
Interstage
Seal Ring
Spacer
Tie Bolts
Interstage Seal
Diaphragm
Case & Stator Assembly
Abradable Shroud
Part Span Shroud




STJ227 COMPRESSOR SECTION ASSEMBLY

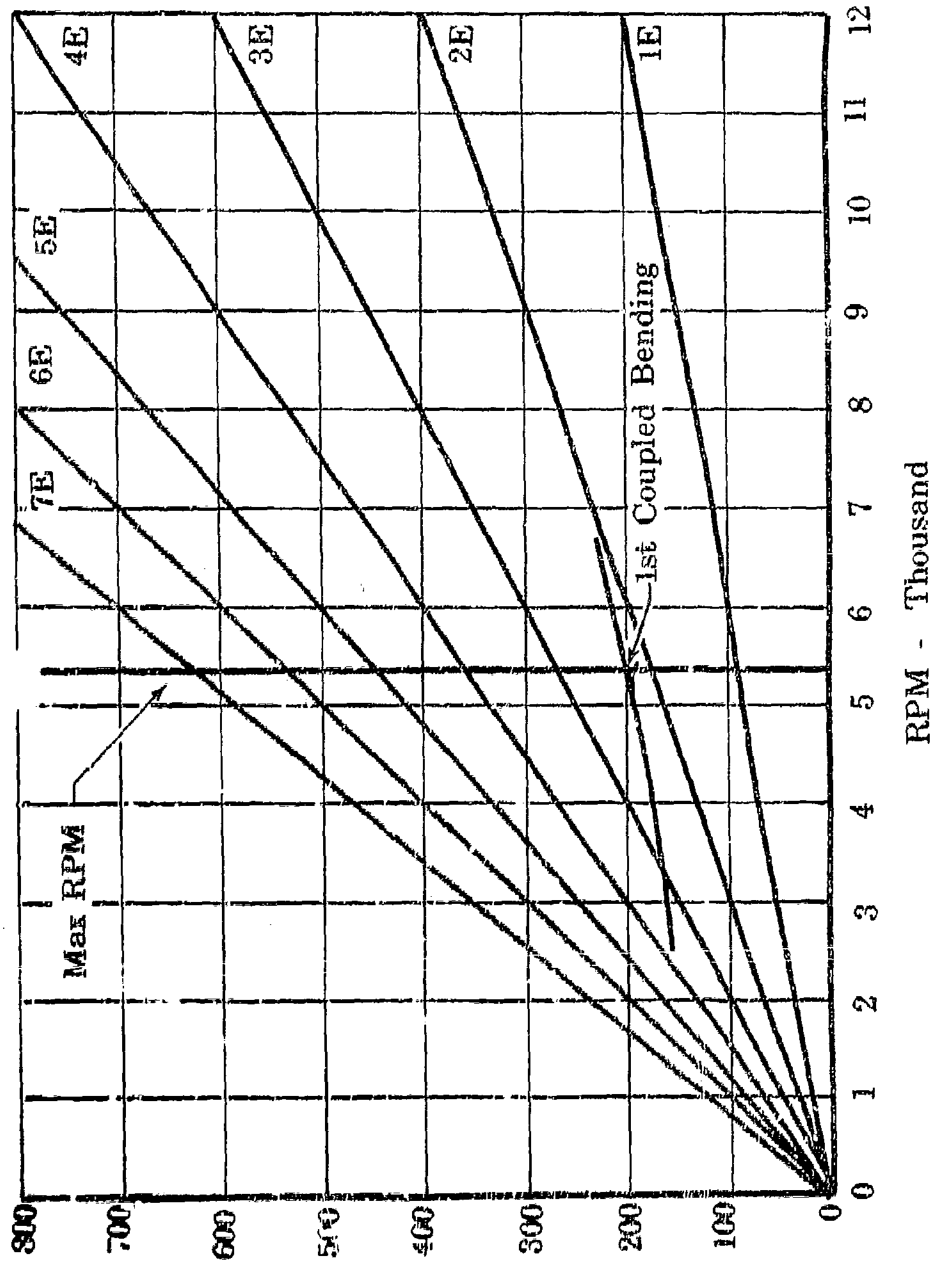



STJ227 COMPRESSOR SECTION ASSEMBLY



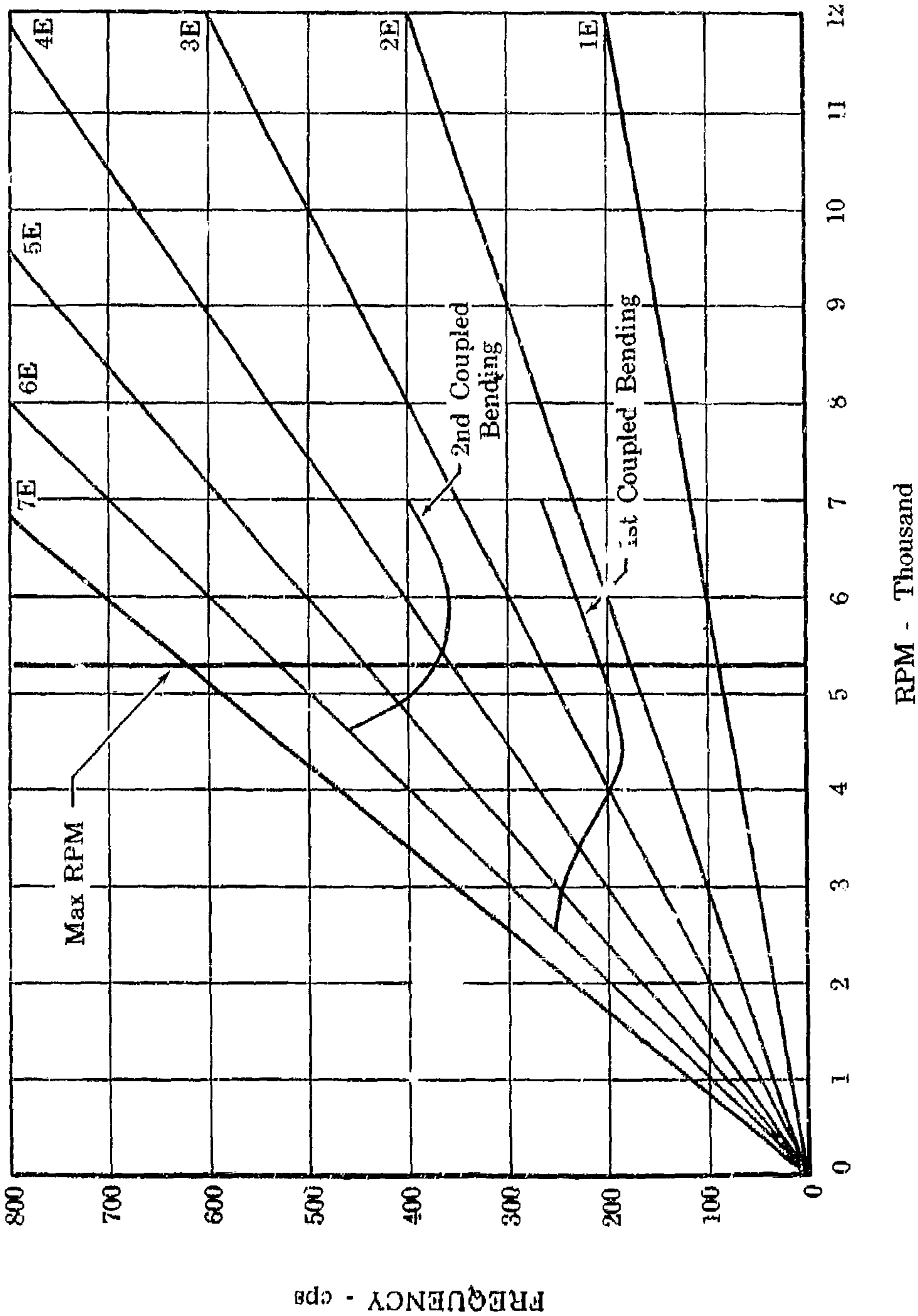

EXTENDED ROOT BLADE WITH MECHANICAL DAMPING

Figure 2B-35



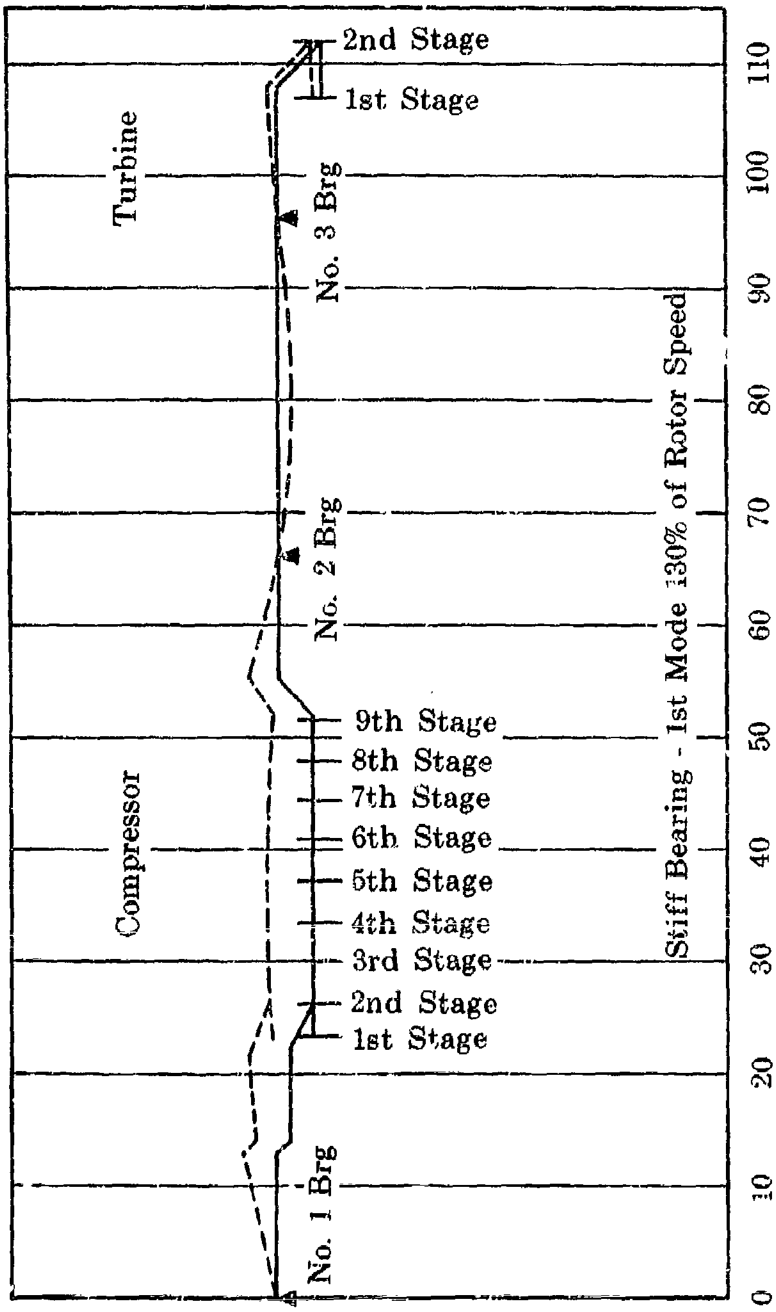

FIRST ROTOR RESONANCE DIAGRAM

Figure 2B-36



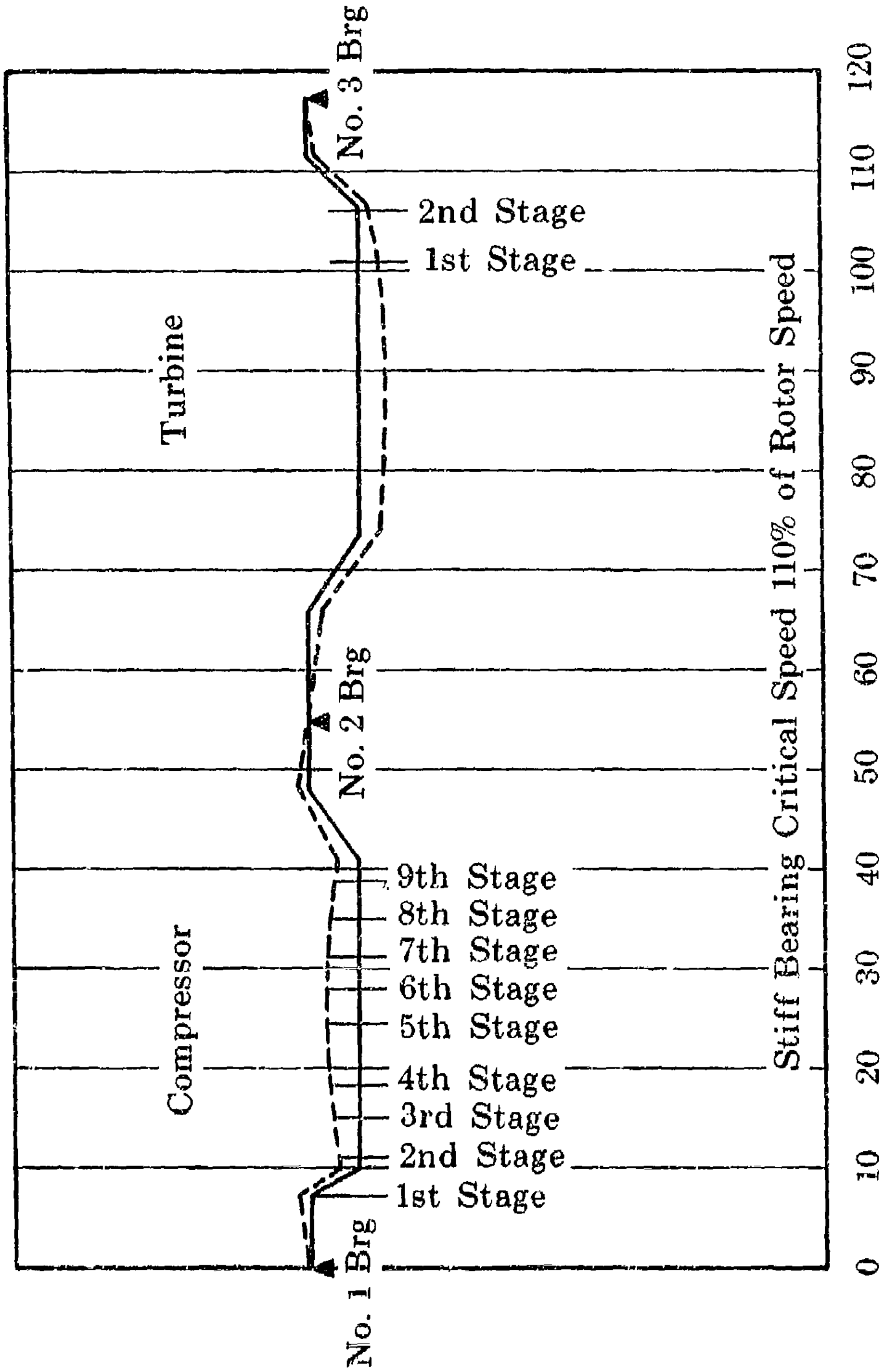

THIRD ROTOR RESONANCE DIAGRAM

Figure 2B-37



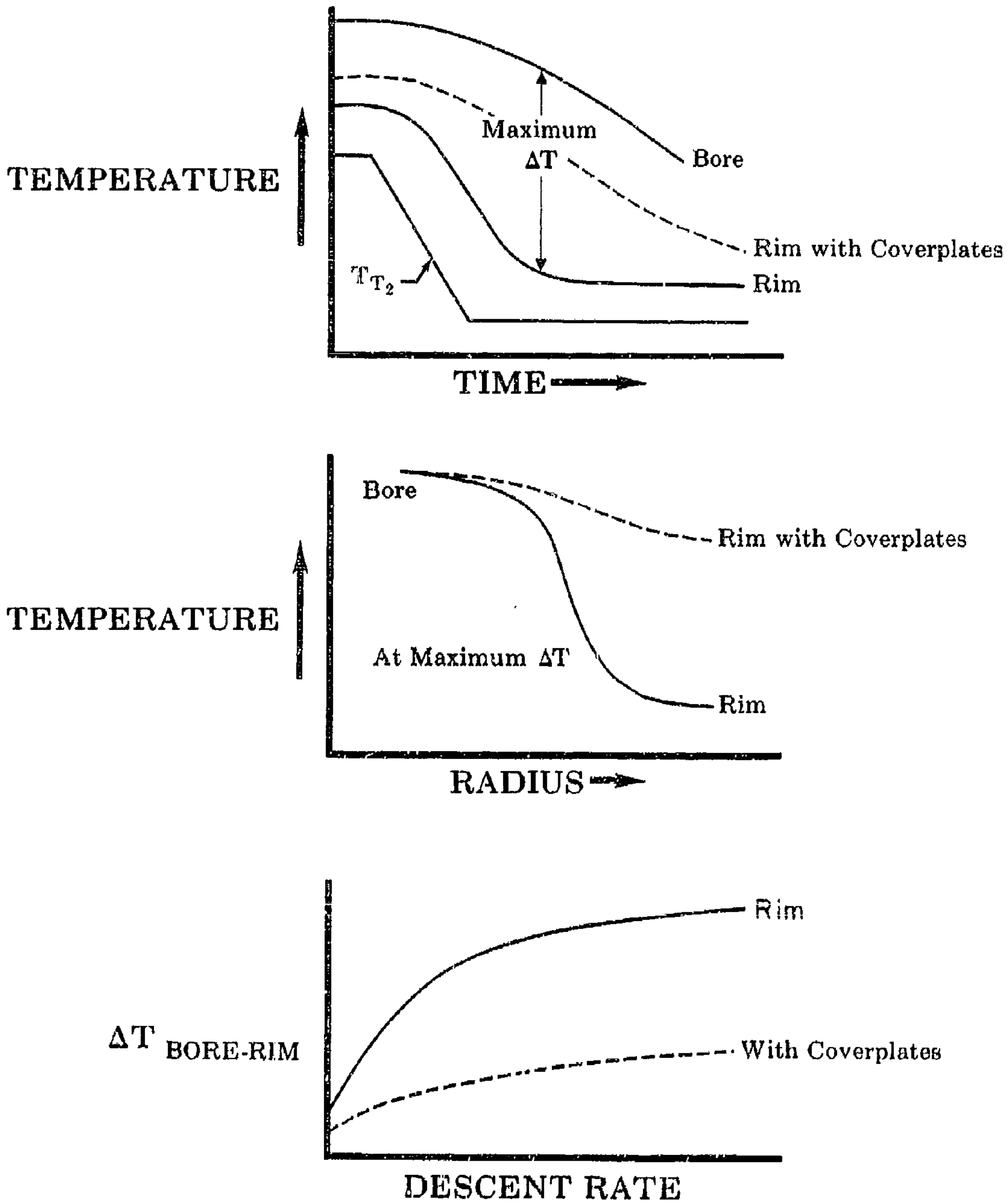

STJ227 ROTOR STIFF BEARING MODE SHAPE

Figure 2B-38



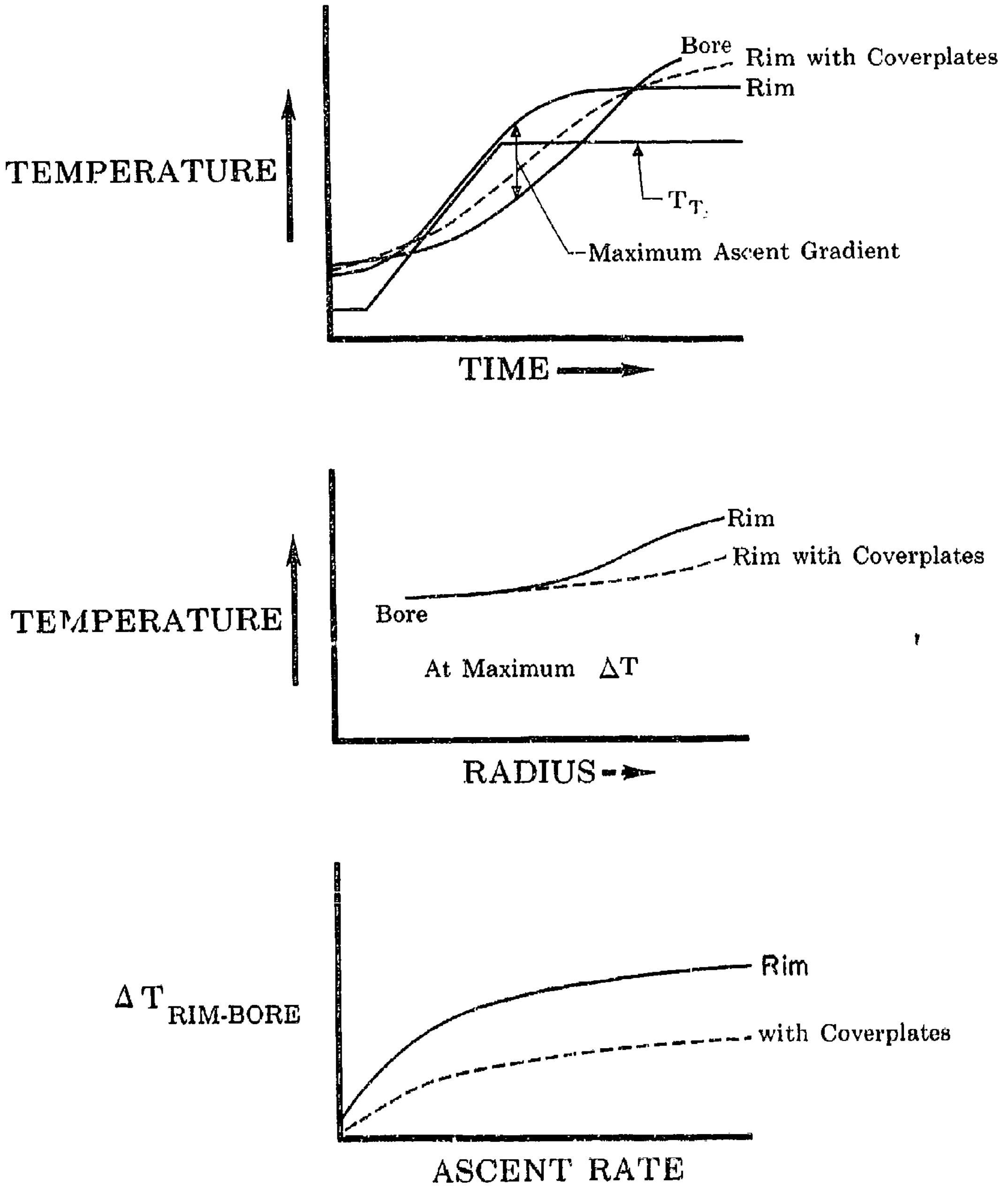

JT11D-20 ROTOR STIFF BEARING MODE SHAPE

Figure 2B-39



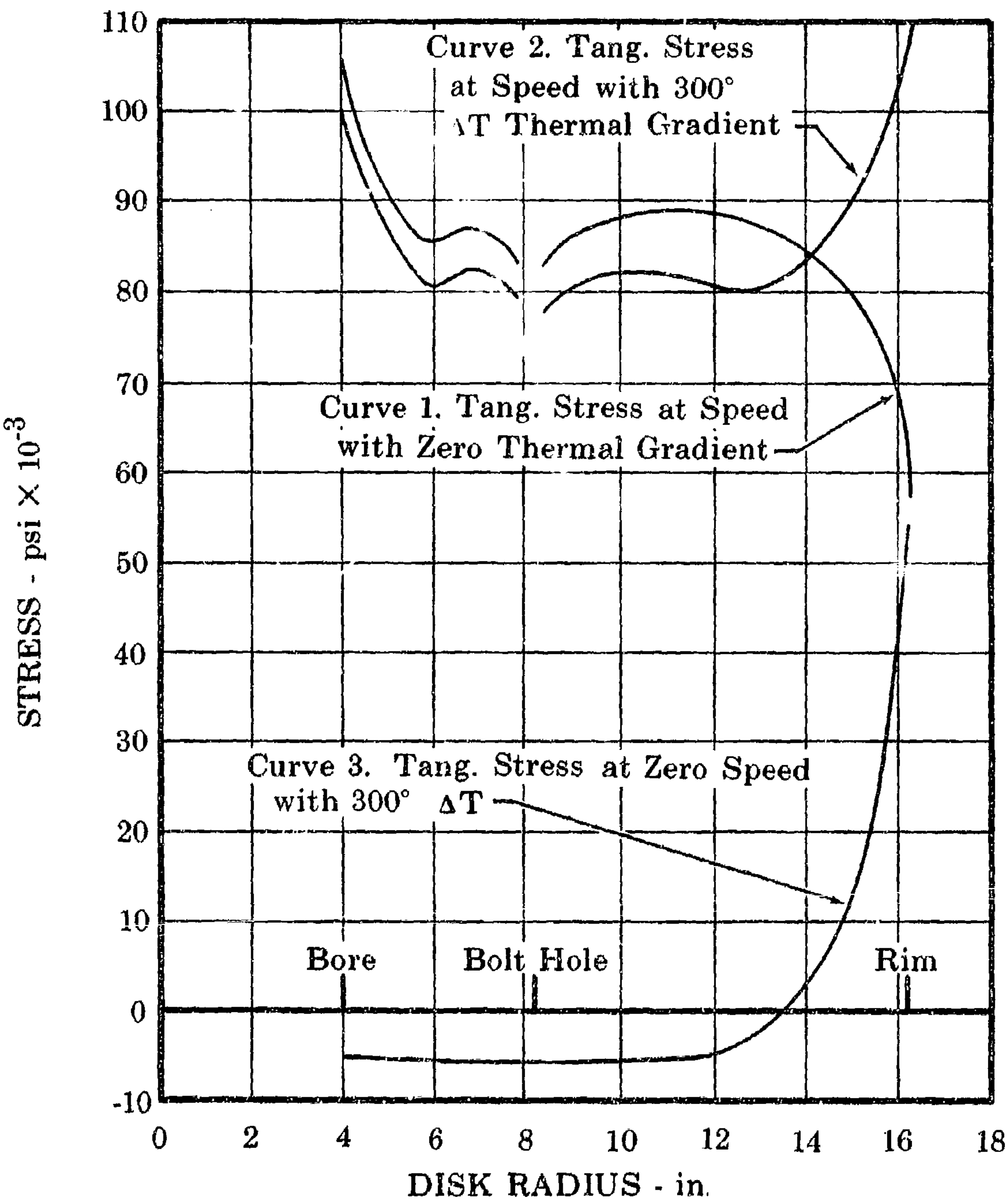

TYPICAL JT11D-20 DESCENT DISK TEMPERATURES

Figure 2B-40



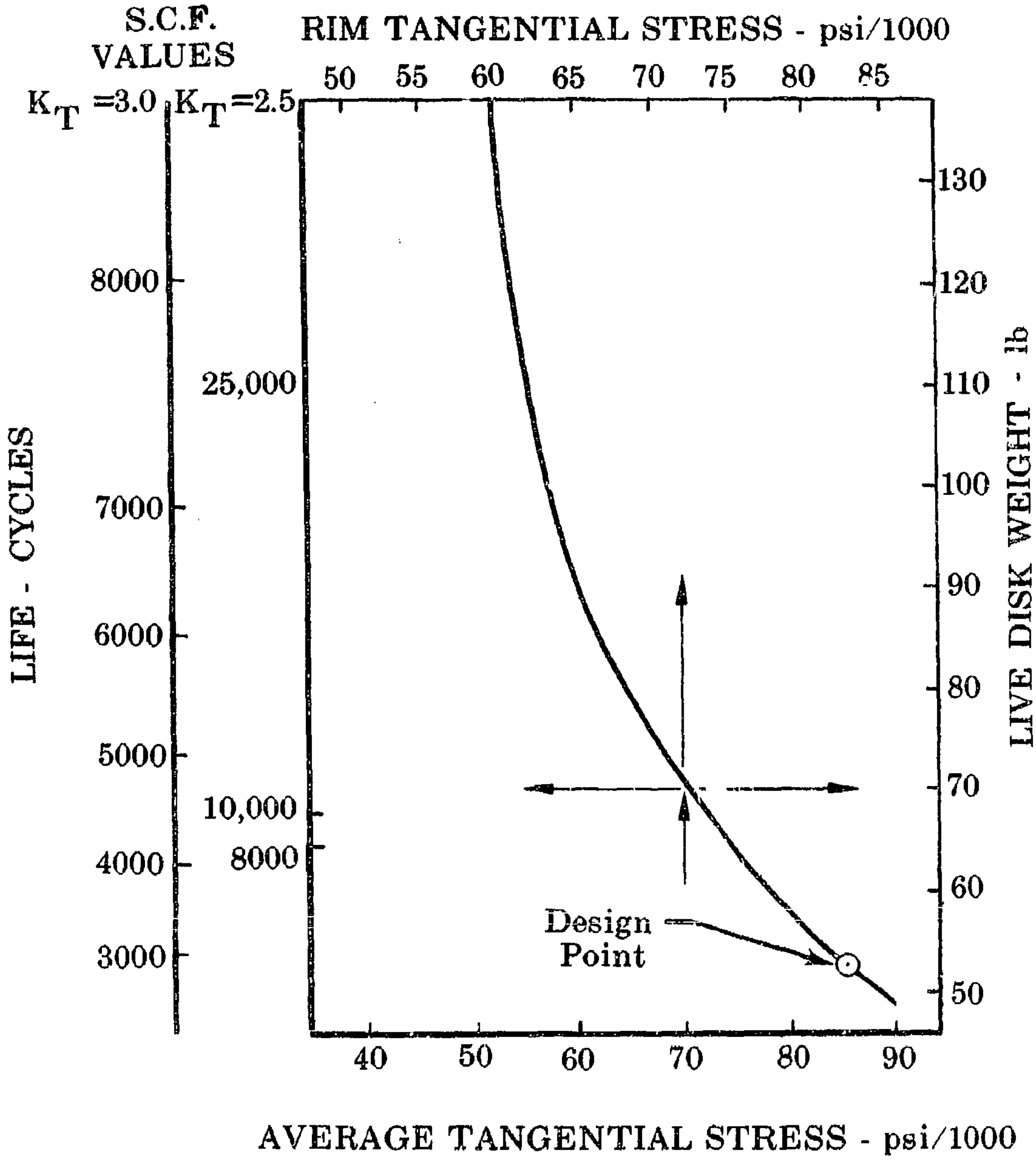

TYPICAL JT11D-20 ASCENT DISK TEMPERATURES

Figure 2B-41



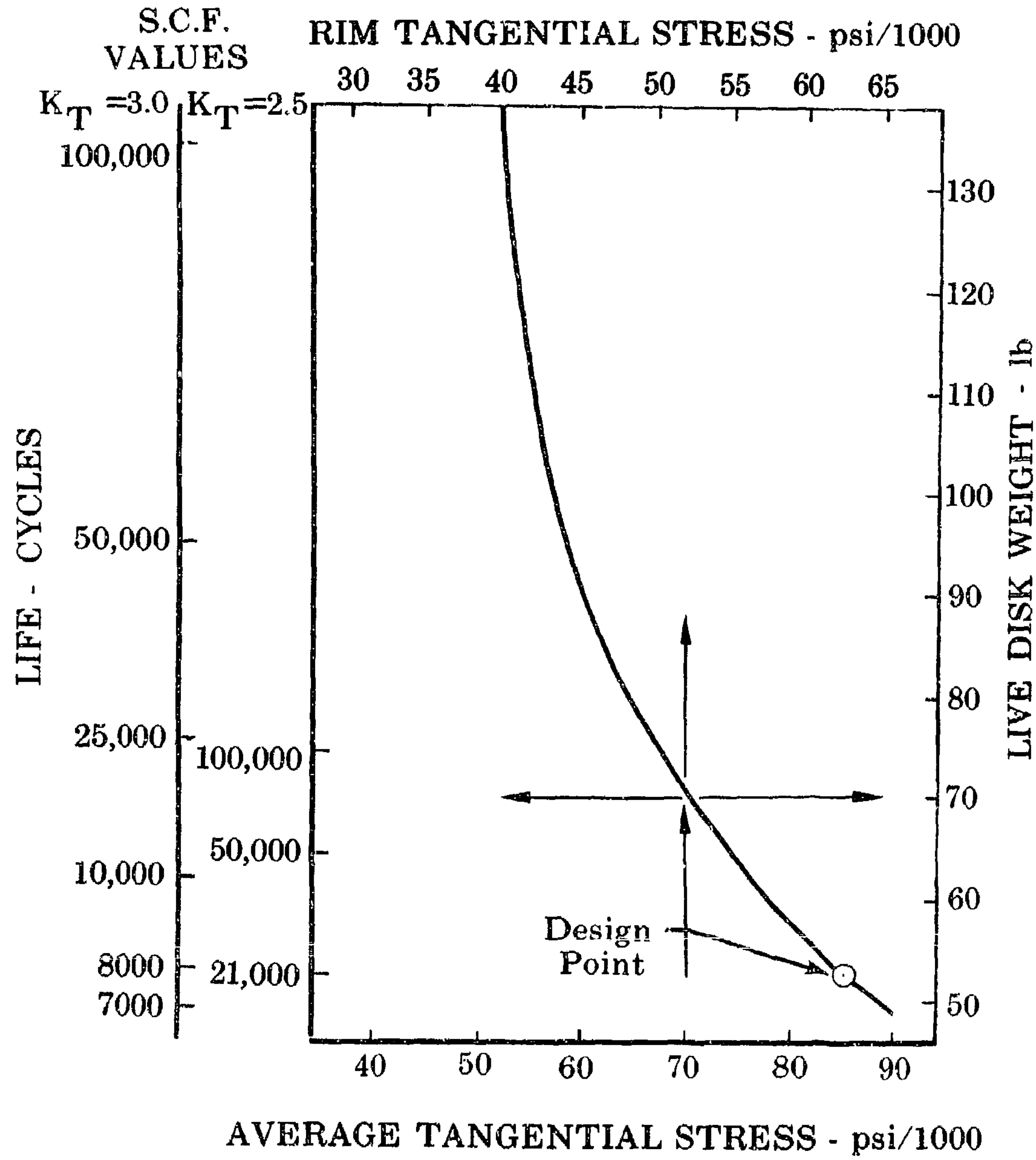

EFFECT OF THERMAL GRADIENTS ON LCF LIFE

Figure 2B-42



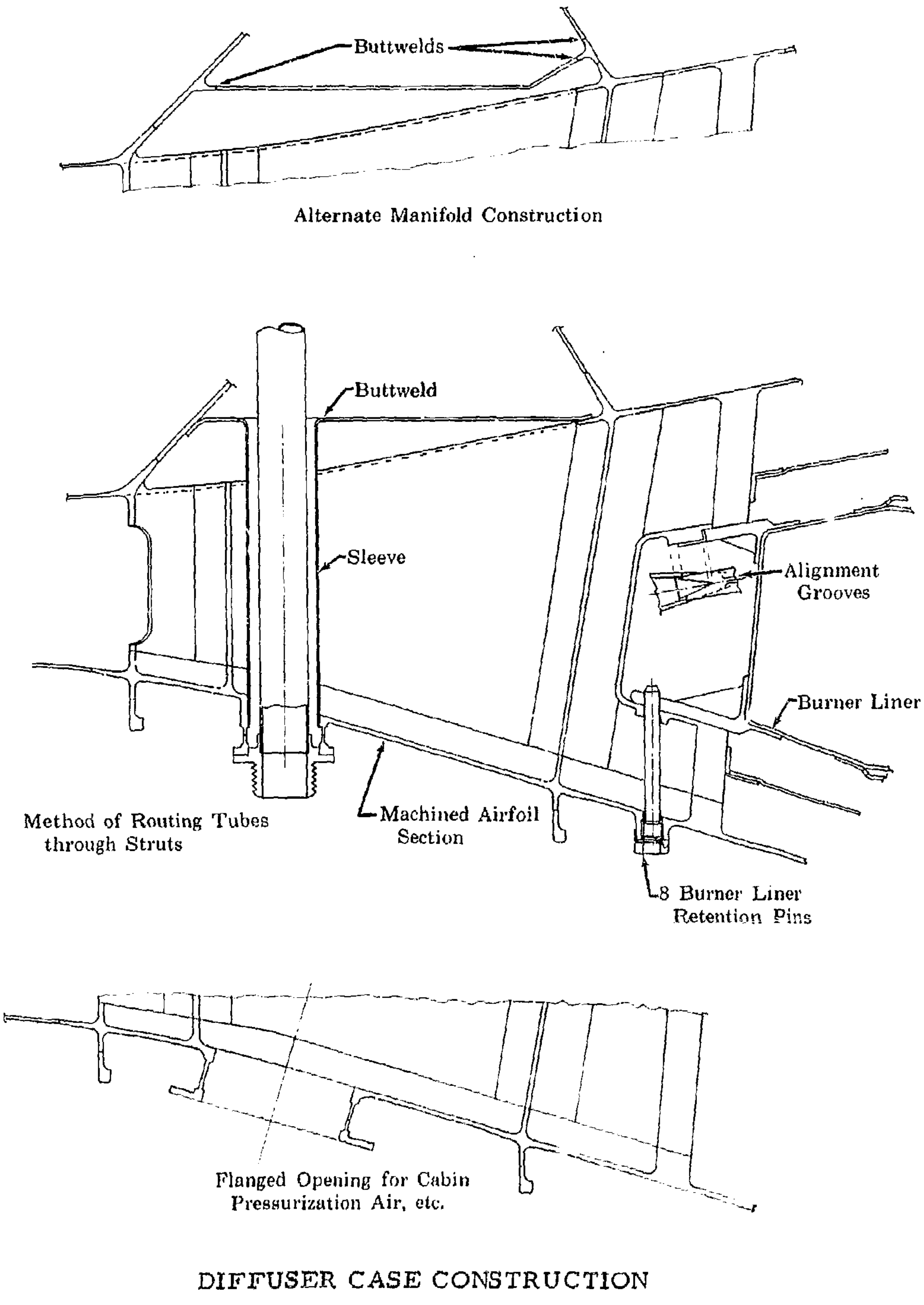

TYPICAL COMPRESSOR DISK THERMAL GRADIENT (250°F △T)

Figure 2B-43



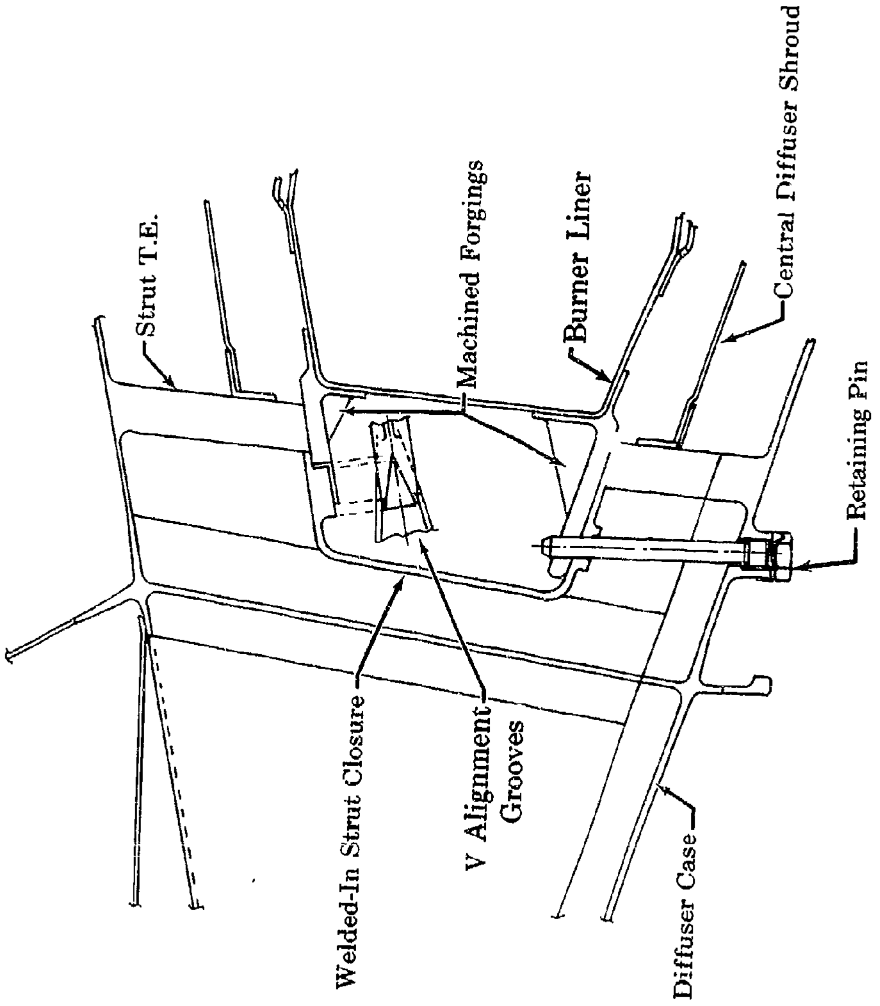

TYPICAL COMPRESSOR DISK THERMAL GRADIENT (130°F $\Delta$T)

Figure 2B-44



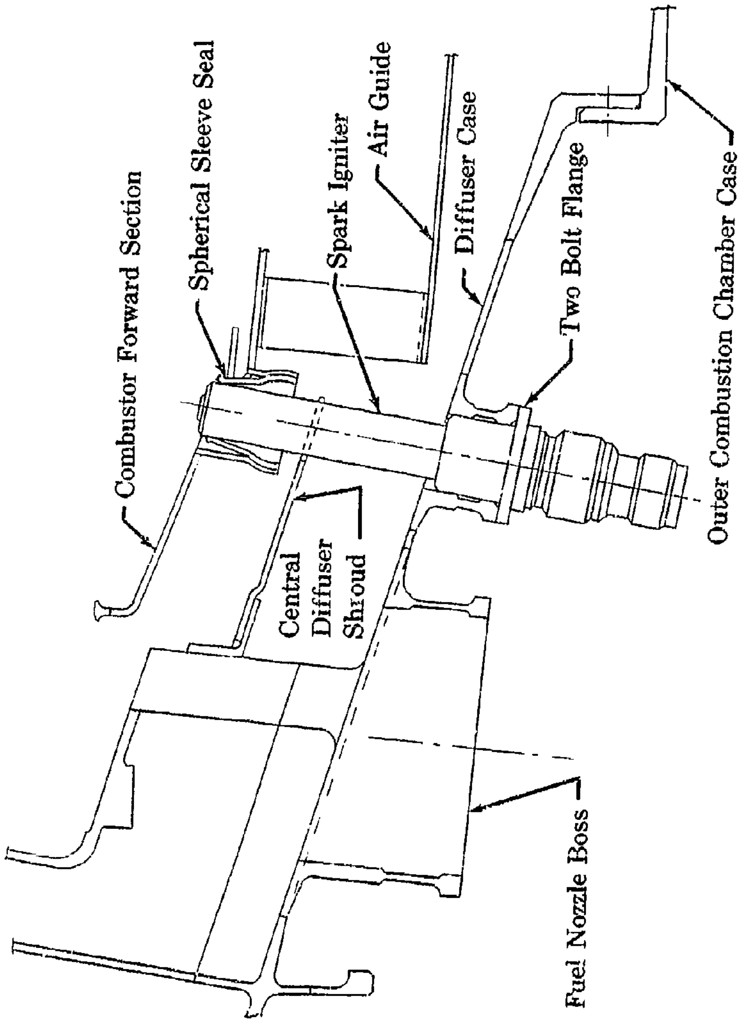

DIFFUSER CASE CONSTRUCTION

Figure 2B-45



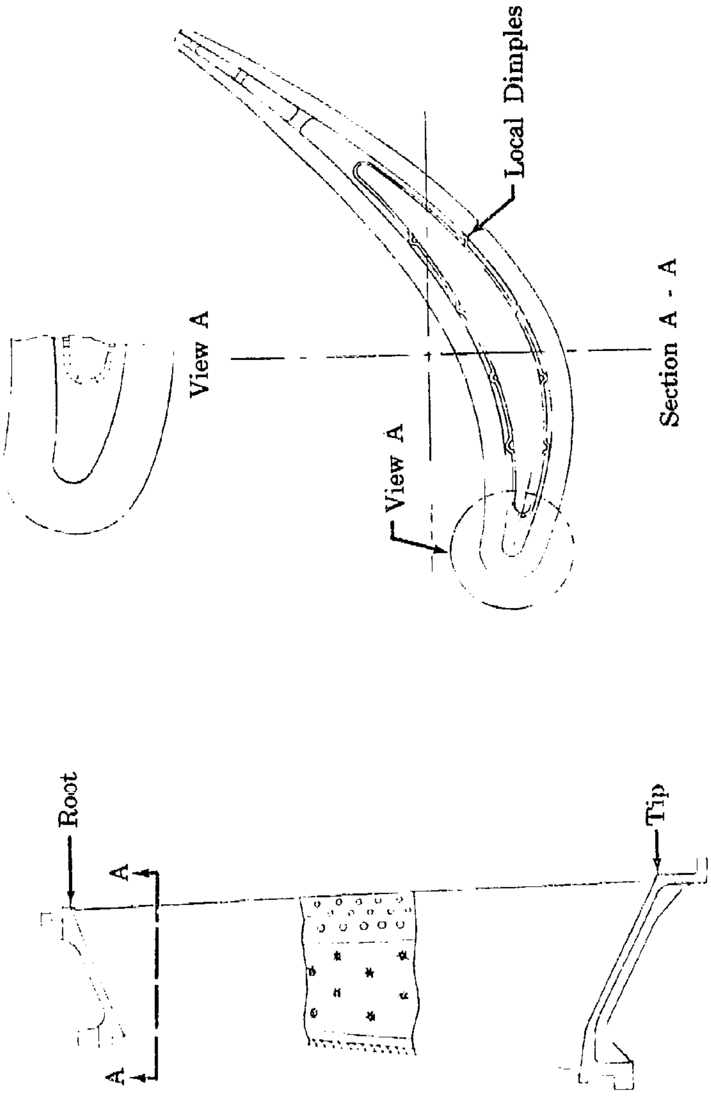

FRONT COMBUSTOR SECTION

Figure 2B-46



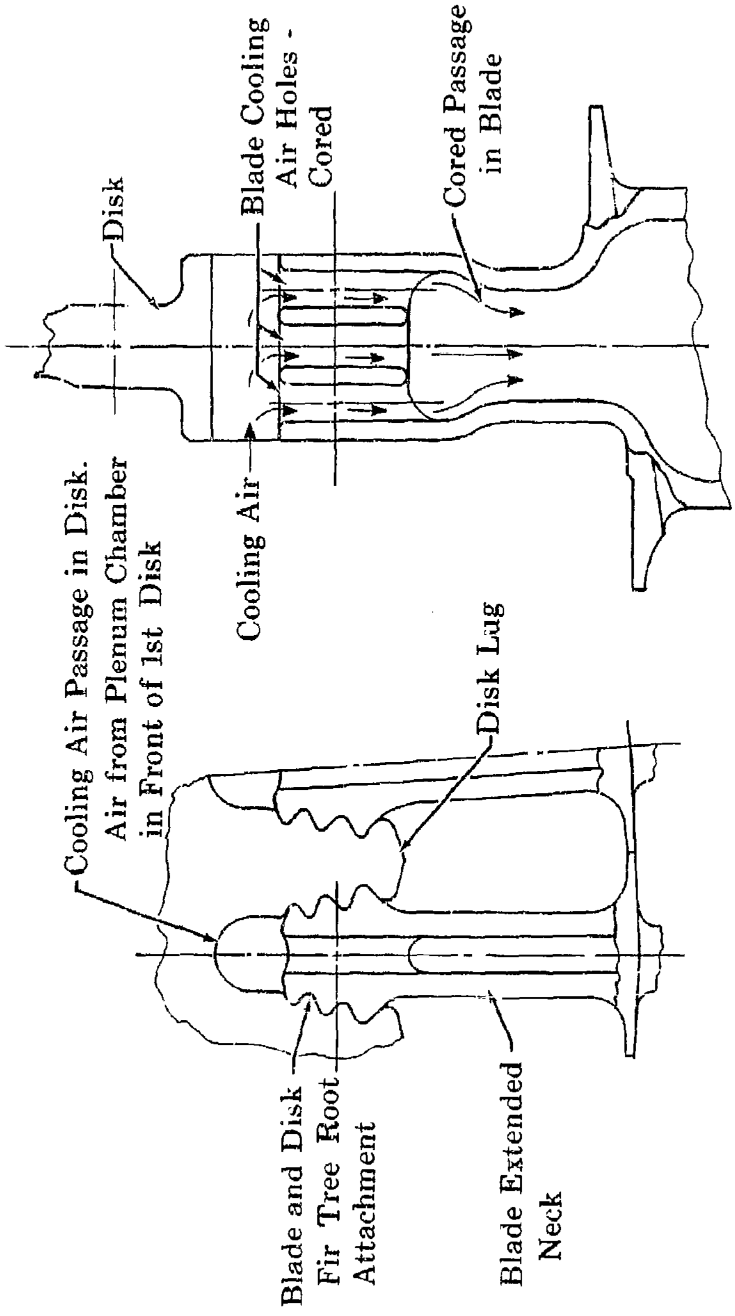

SPARK IGNITORS

Figure 2B-47



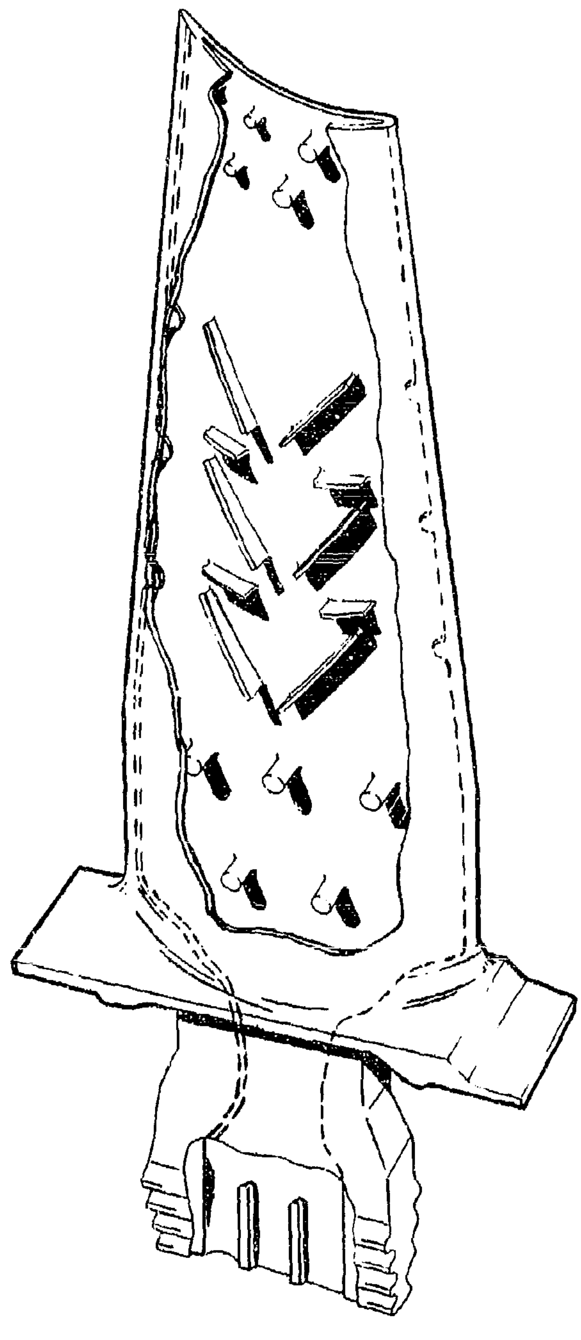

SECOND STAGE TURBINE VANE

Figure 2B-48



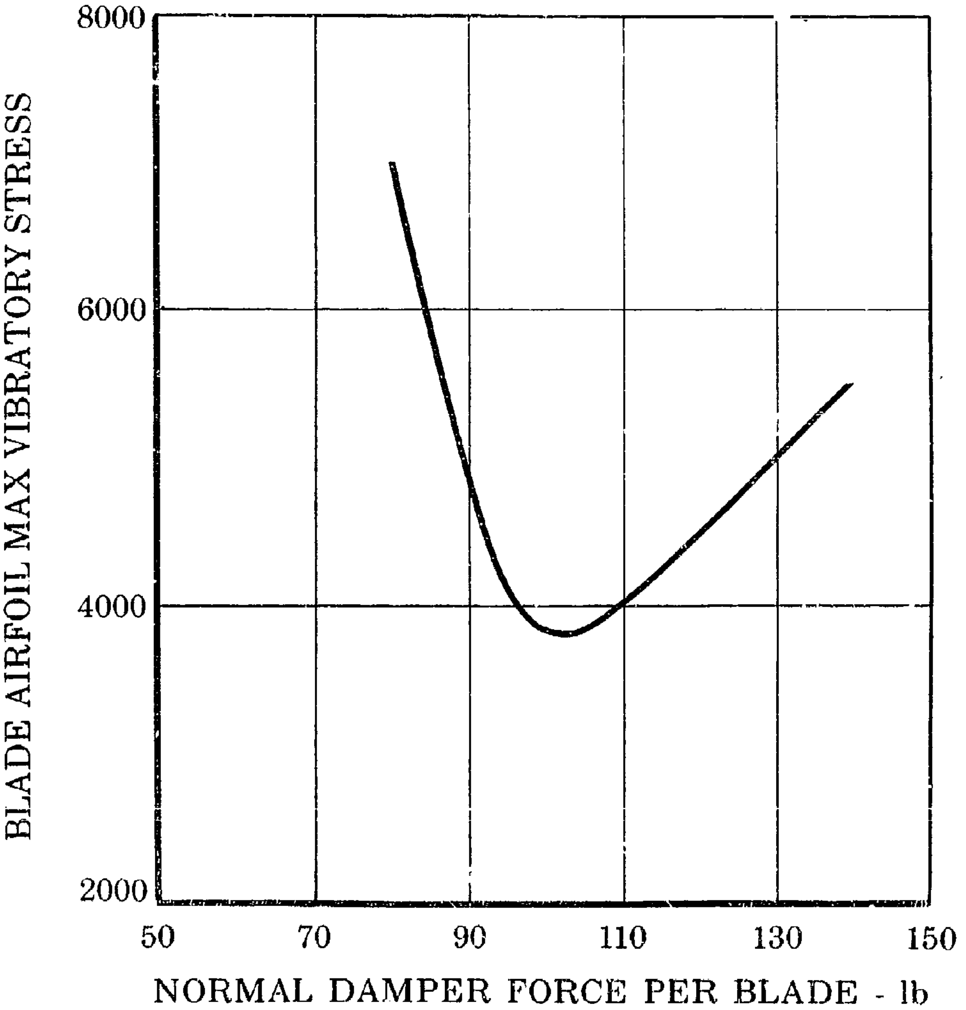

FIRST STAGE BLADE AND DISK COOLING PASSAGES

Figure 2B-49



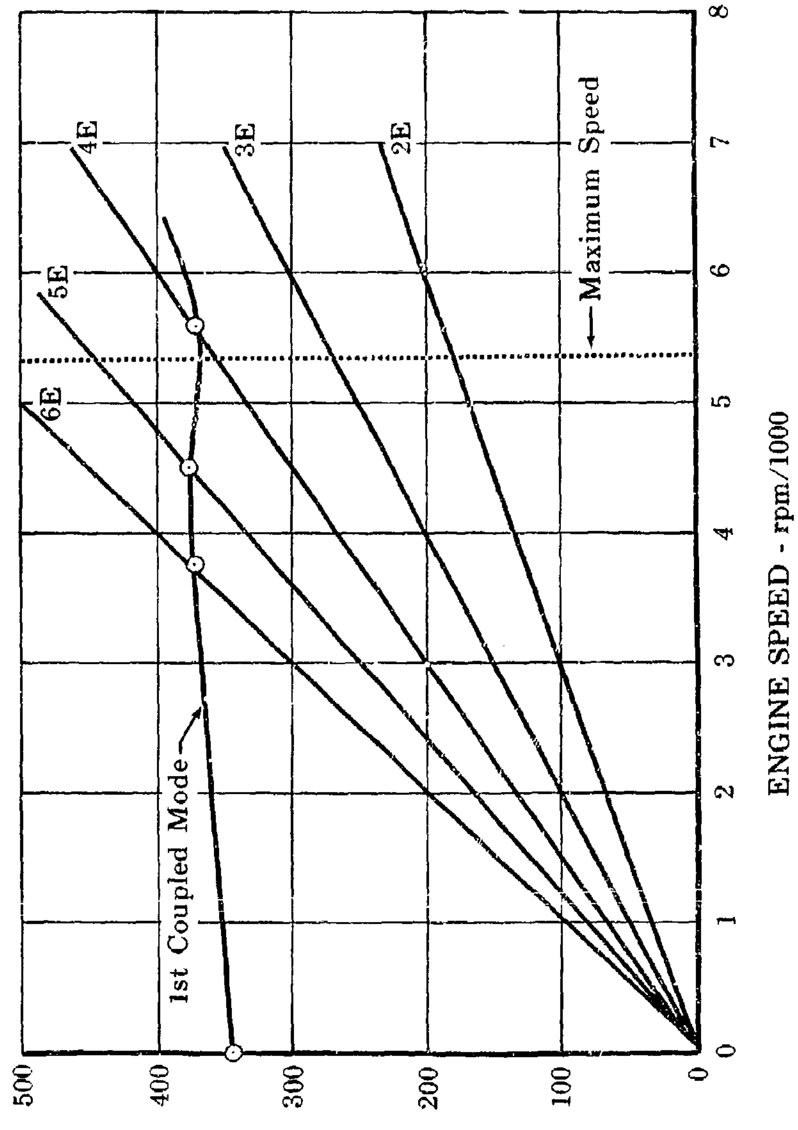
FIRST STAGE BLADE COOLING SCHEME

Figure 2B-50



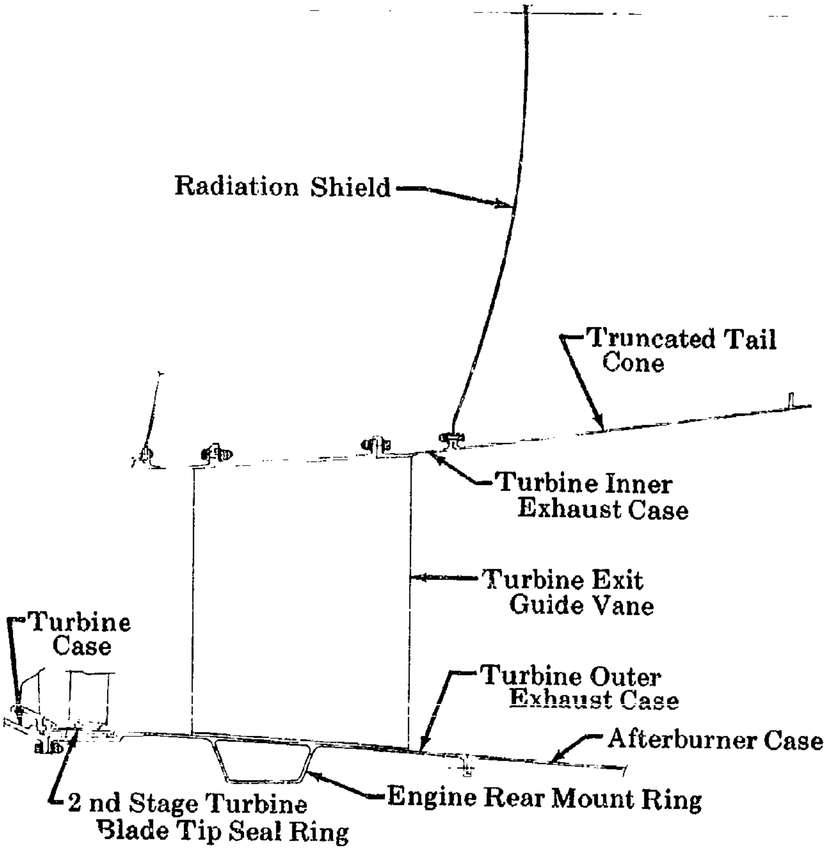

JT11D-20 FIRST STAGE BLADE AIRFOIL STRESS
VS. DAMPER FORCE

Figure 2B-51



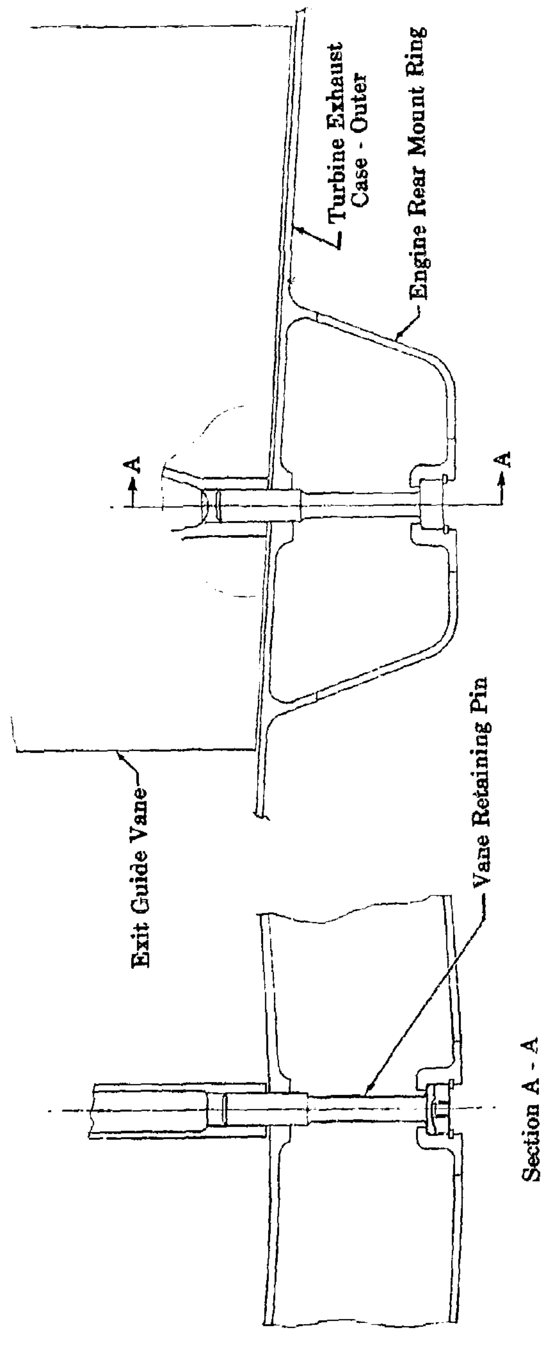

STJ227 FIRST STAGE BLADED-DISK COUPLED MODE

Figure 2B-52



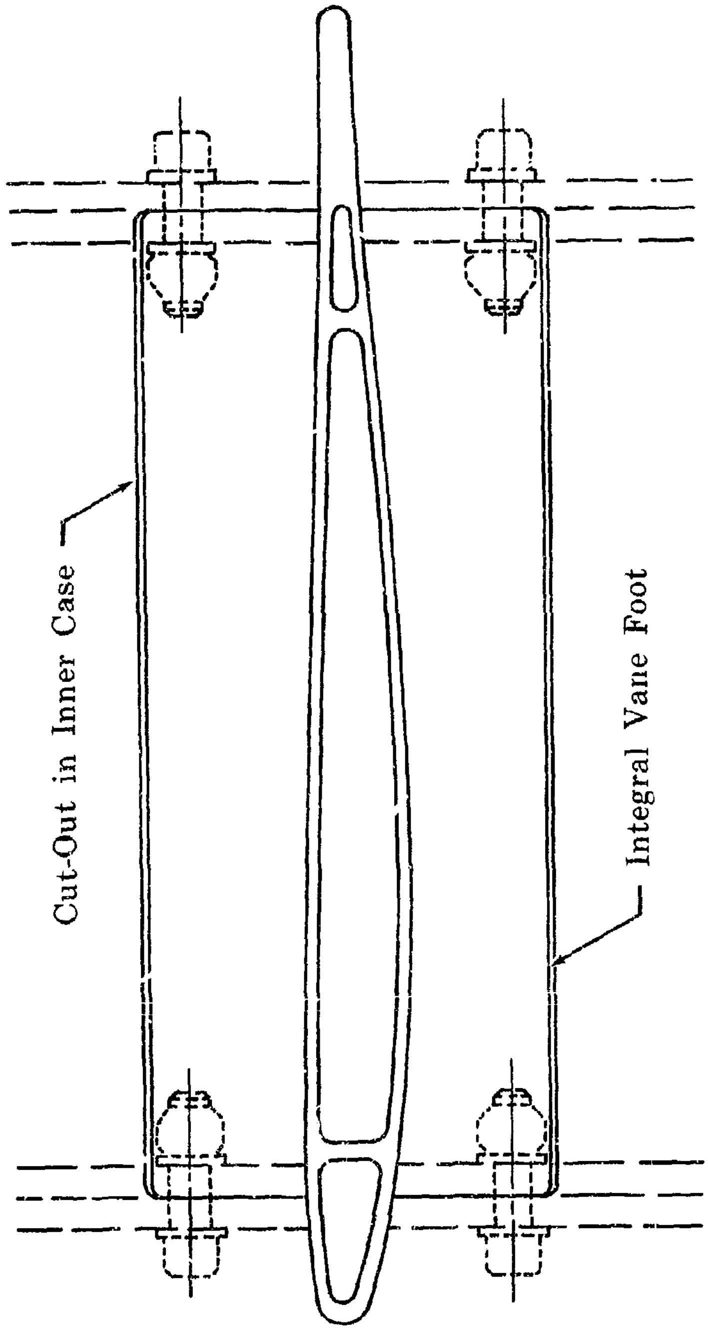

TURBINE EXHAUST SECTION

Figure 2B-53



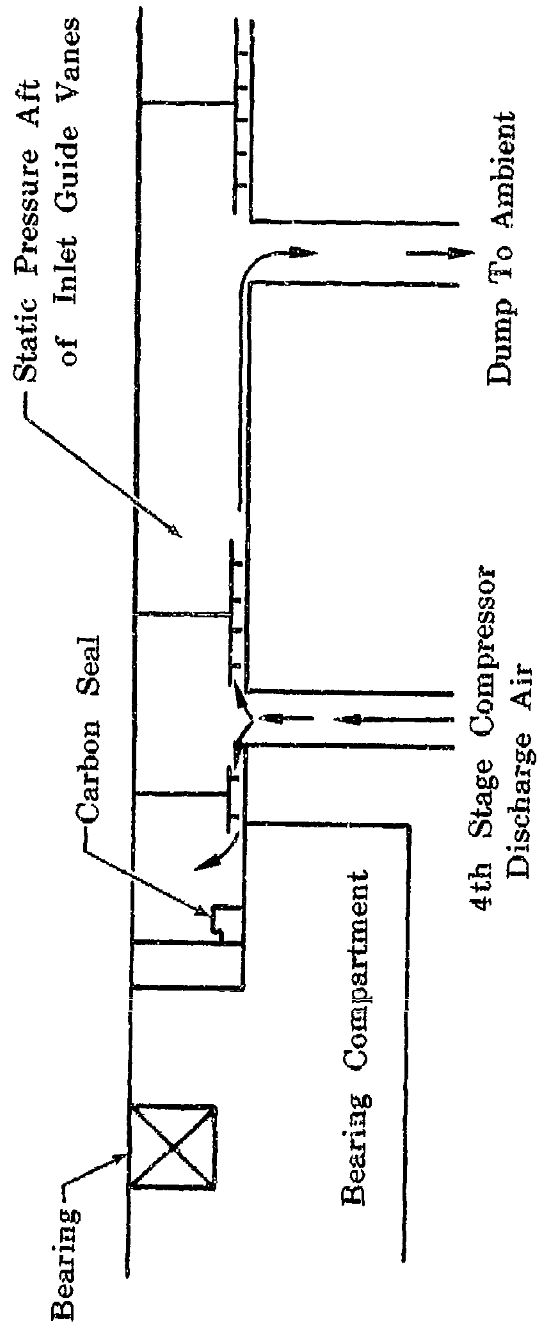

TURBINE EXIT GUIDE VANE OUTER ATTACHMENT

Figure 2B-54



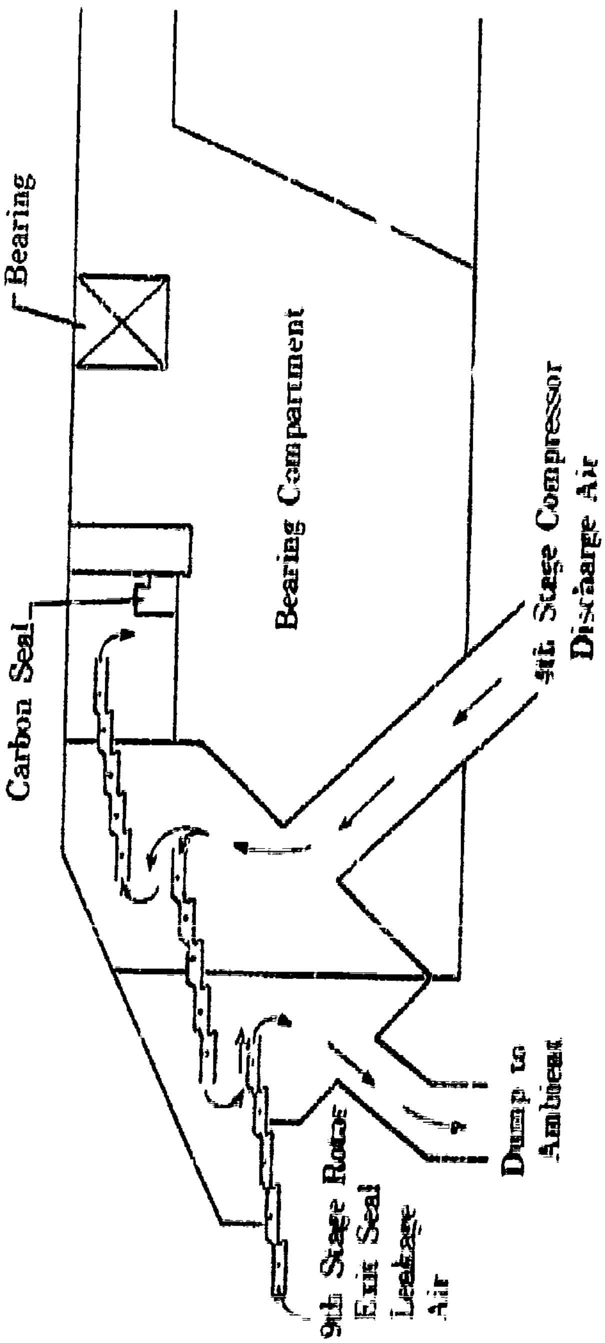

TURBINE EXIT GUIDE VANE ATTACHMENT TO INNER CASE

Figure 2B-55



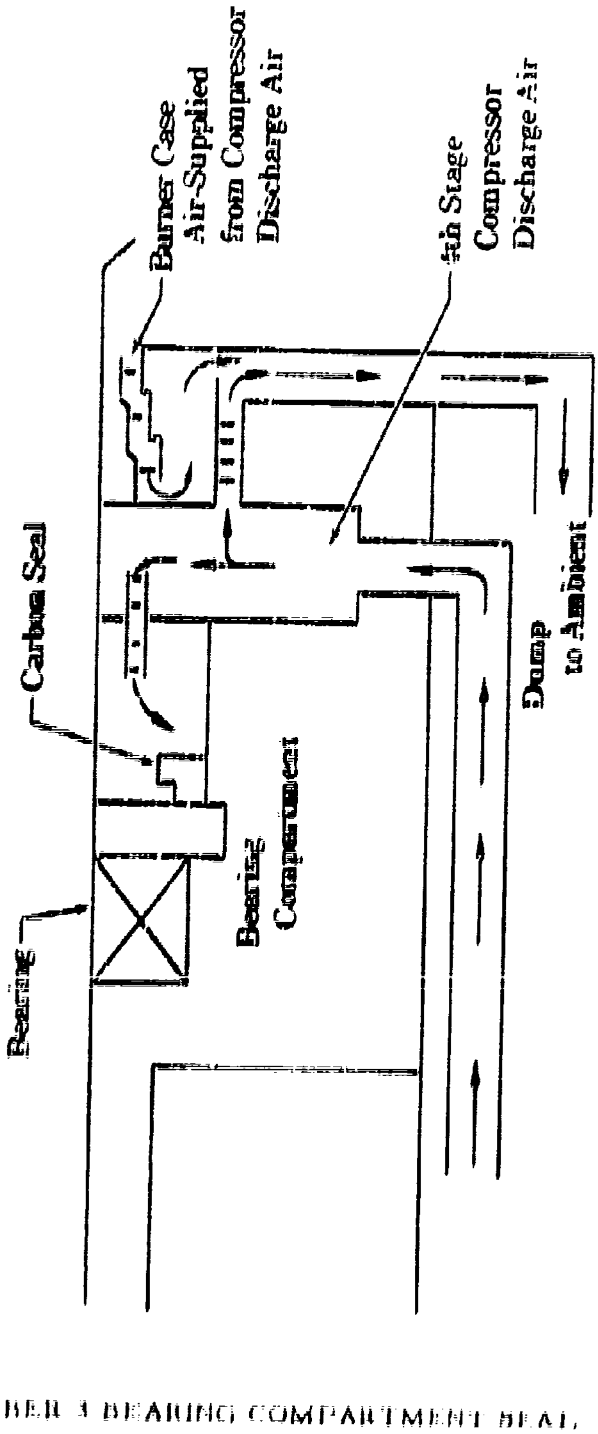

NUMBER 1 BEARING COMPARTMENT SEAL

Figure 2B-56



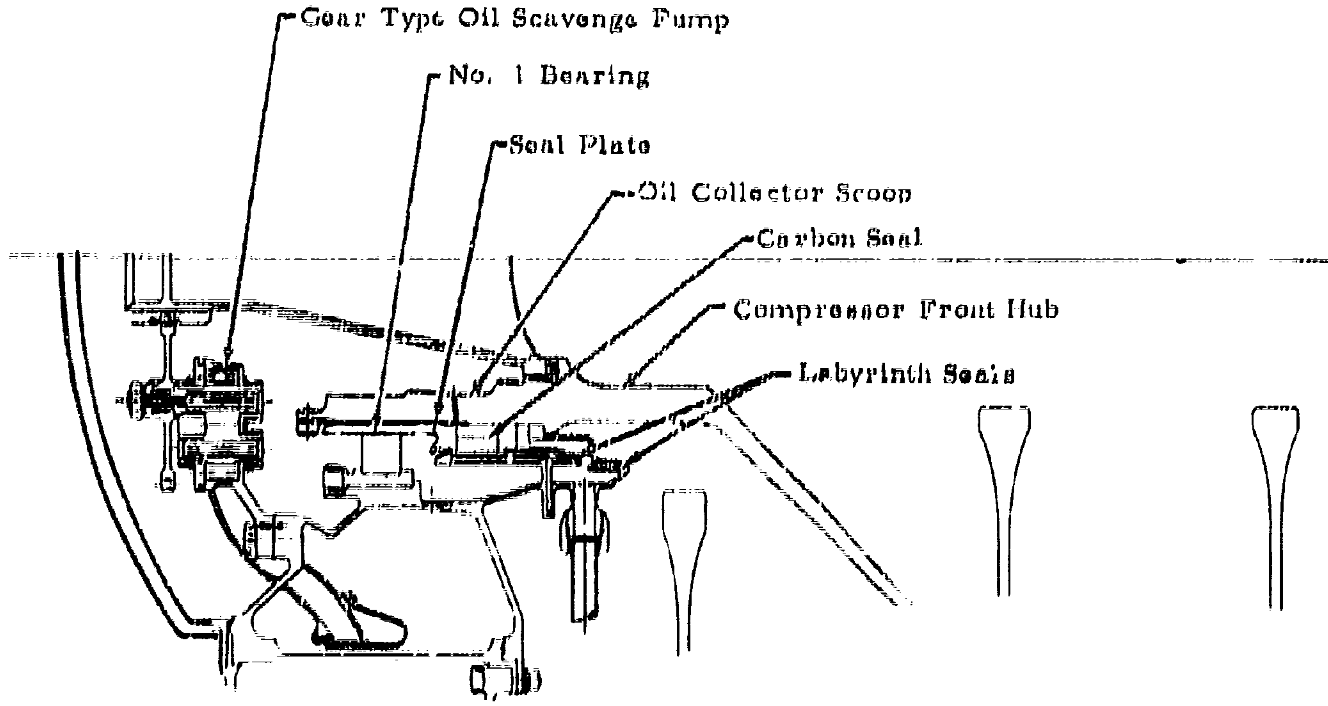

NUMBER 2 BEARING COMPARTMENT SEAL

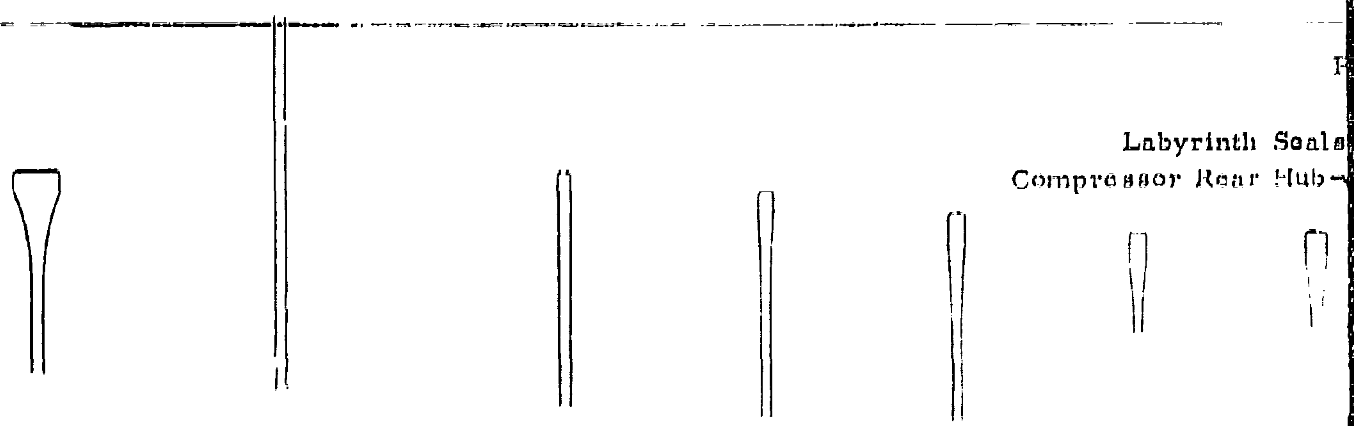
Figure 23-57

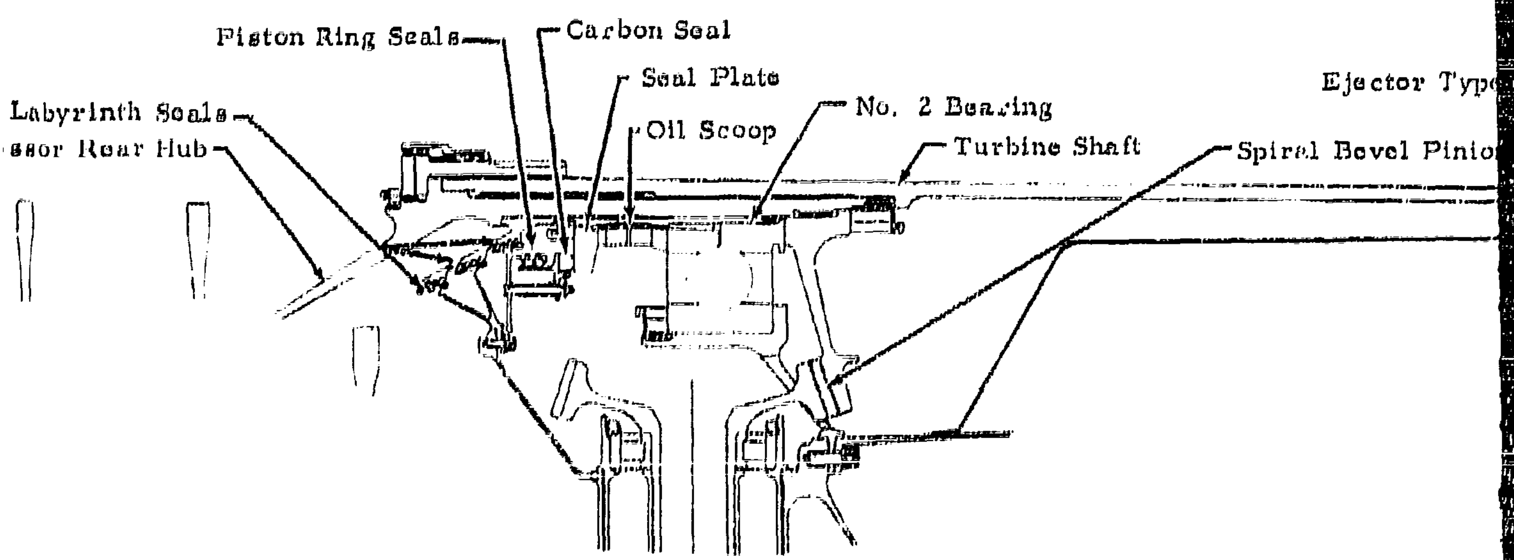



NUMBER 3 BEARING COMPARTMENT SEAL

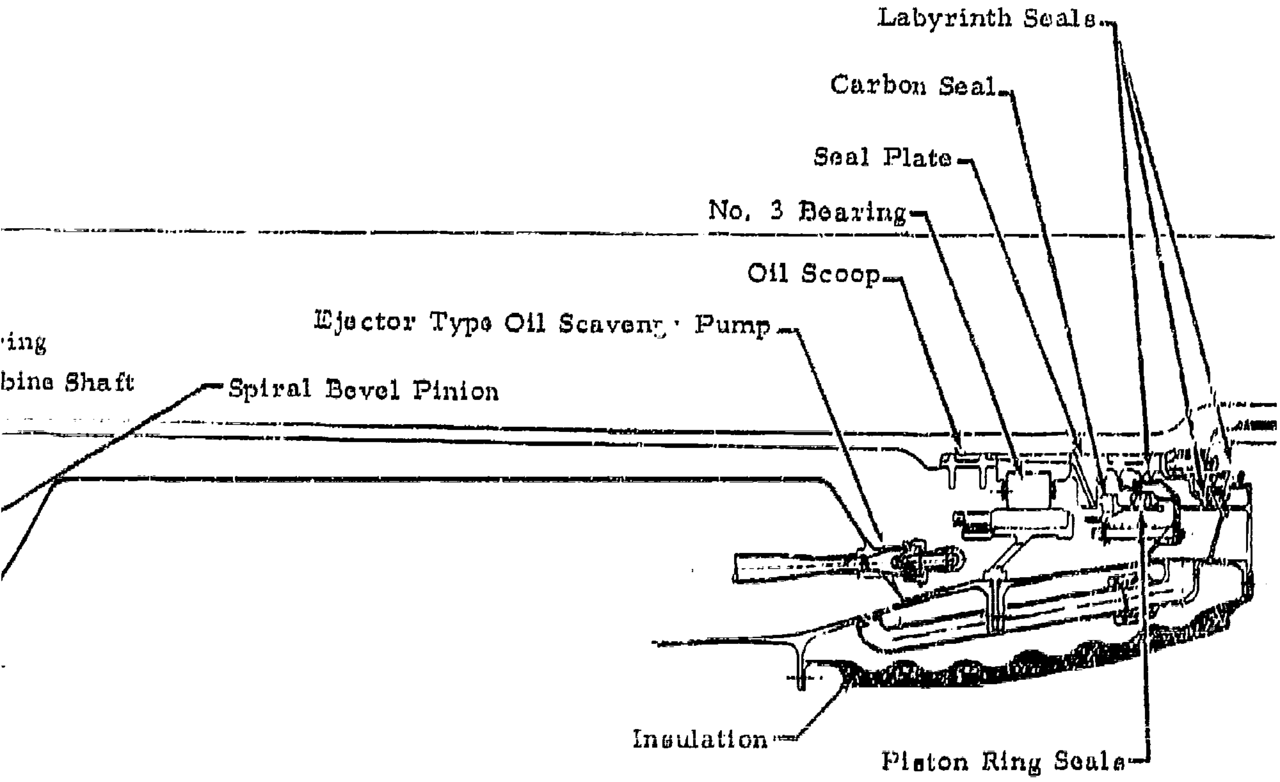
Figure 2B-58


Gear Type Oil Scavenge Pump
No. 1 Bearing
Seal Plate
Oil Collector Scoop
Carbon Seal
Compressor Front Hub
Labyrinth Seals


Labyrinth Seals
Compressor Rear Hub

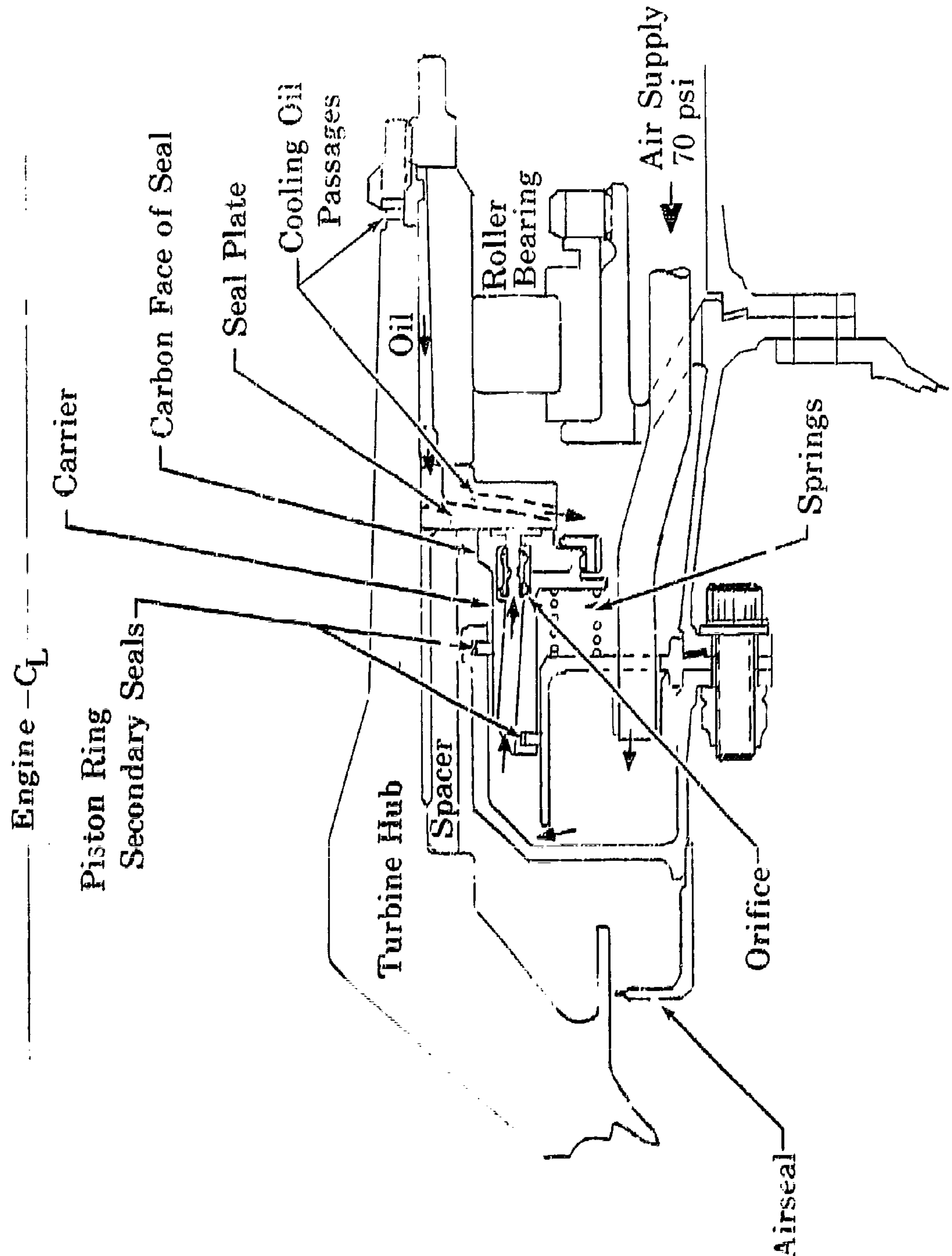


Piston Ring Seals
Carbon Seal
Seal Plate
Labyrinth Seals
No. 2 Bearing
Oil Scoop
ssor Rear Hub
Turbine Shaft
Spiral Bevel Pinio
Ejector Typ



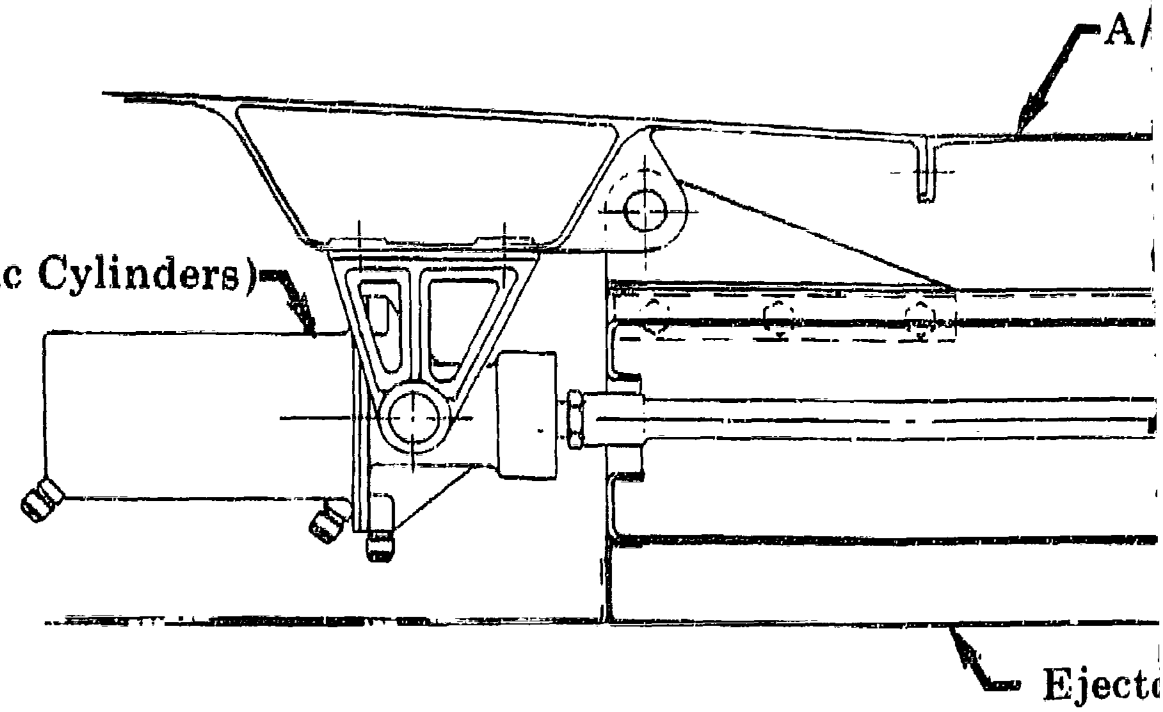

BEARING COMPARTMENTS

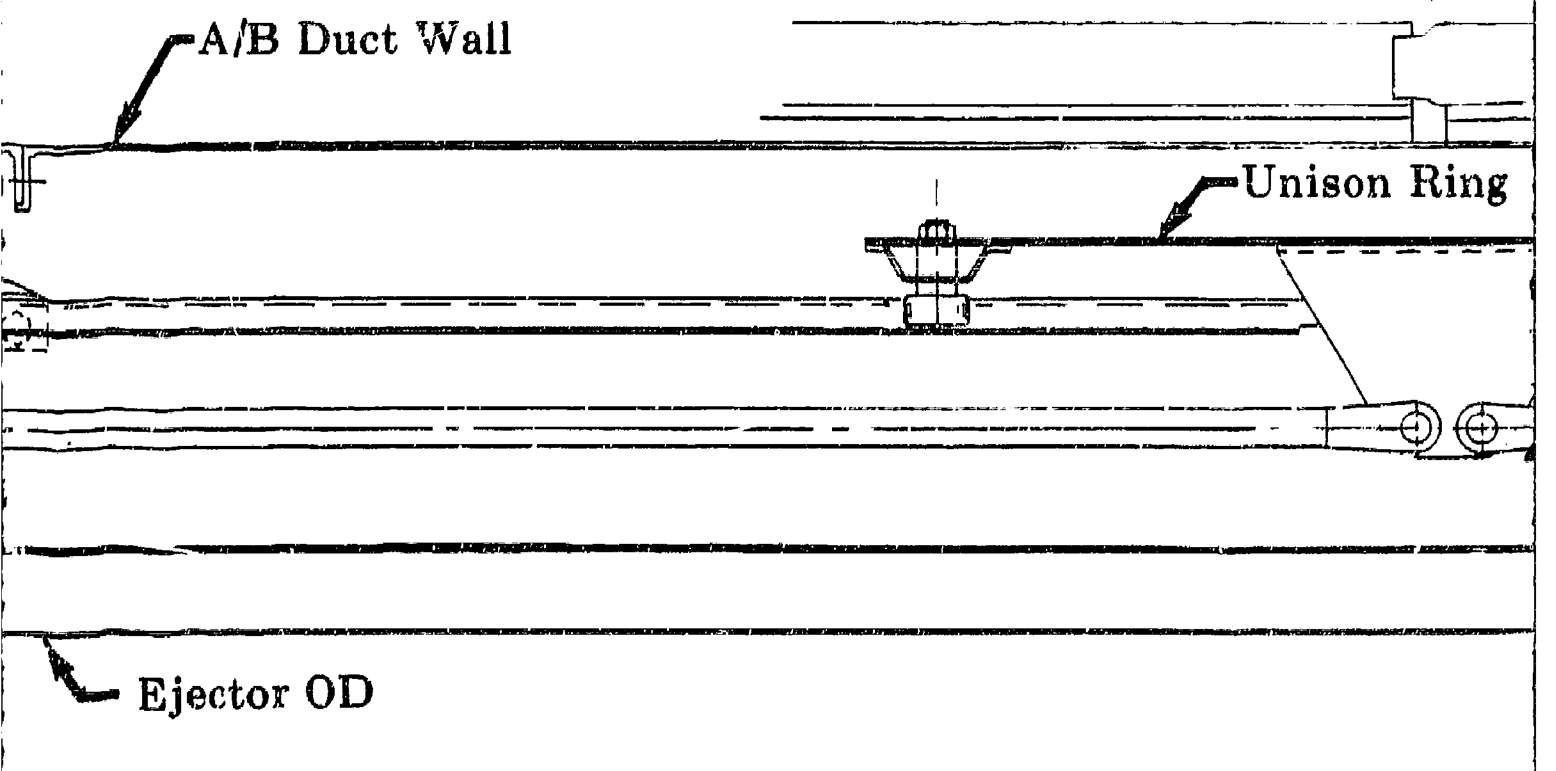
Figure 2B-59



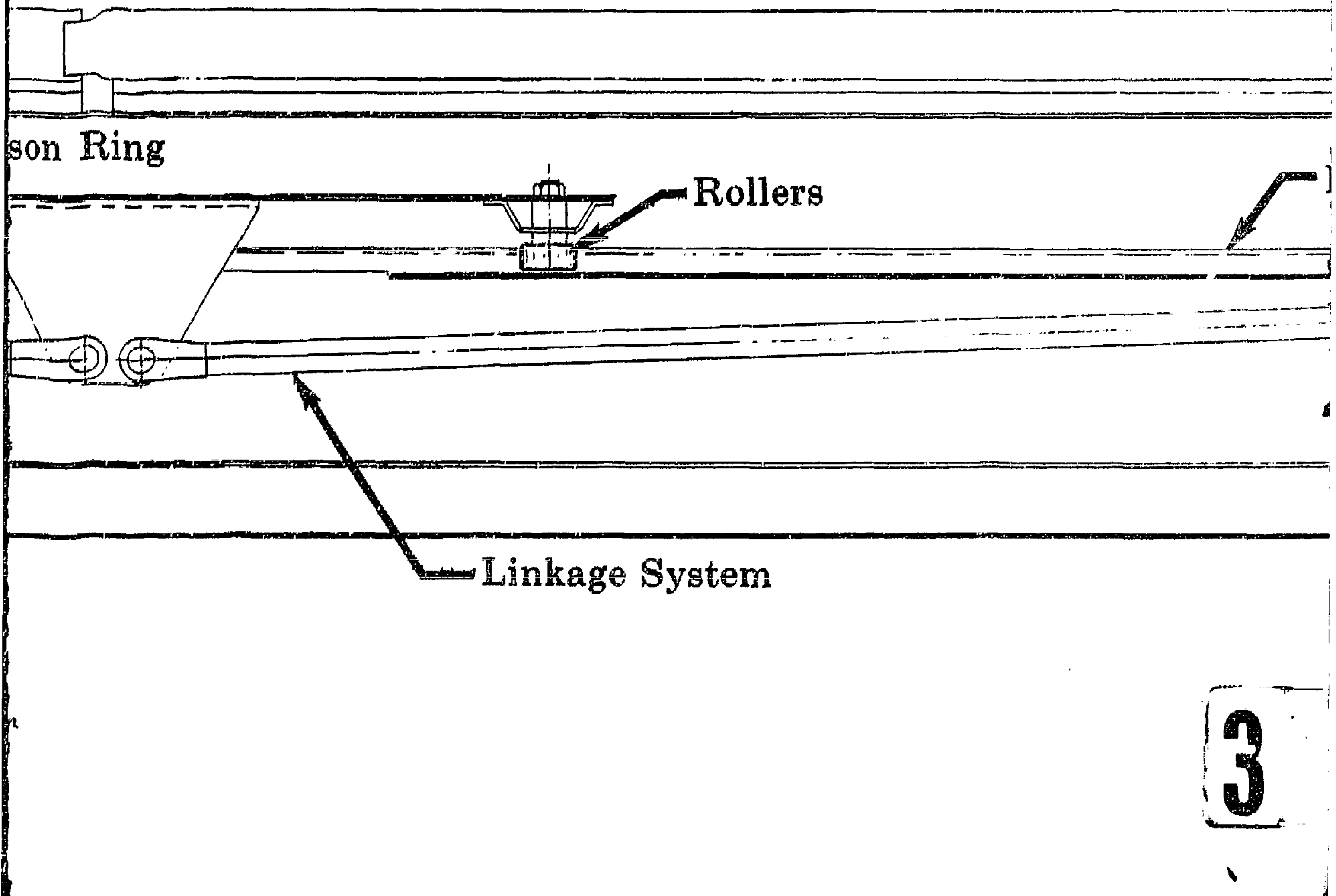

HYDROSTATIC SEAL ARRANGEMENT

Figure 2B-60

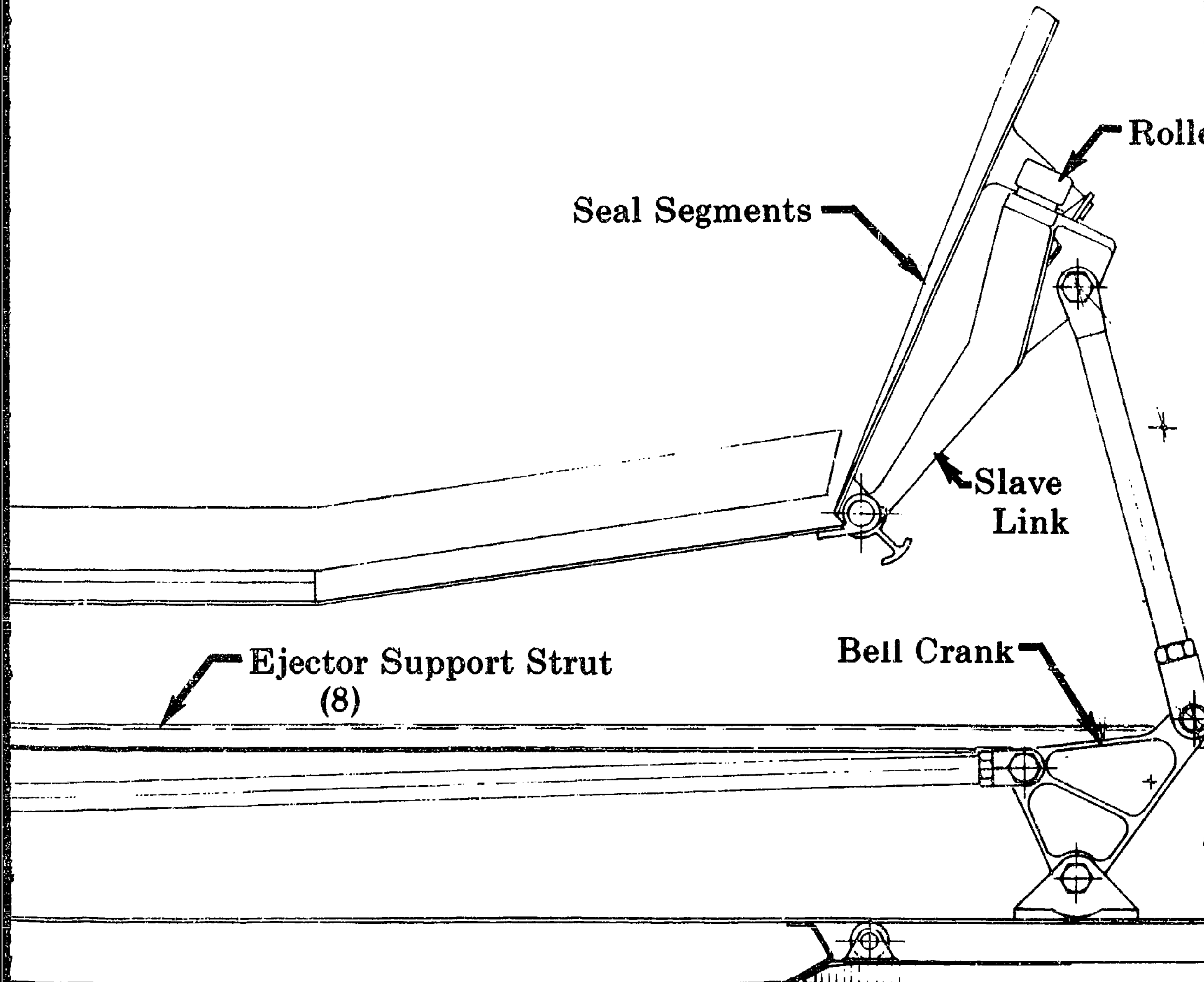

A/
(Hydraulic Cylinders)
8
Eject




2



son Ring
Rollers
Linkage System
3



Rolle
Seal Segments
Slave
Link
Ejector Support Strut
(8)
Bell Crank

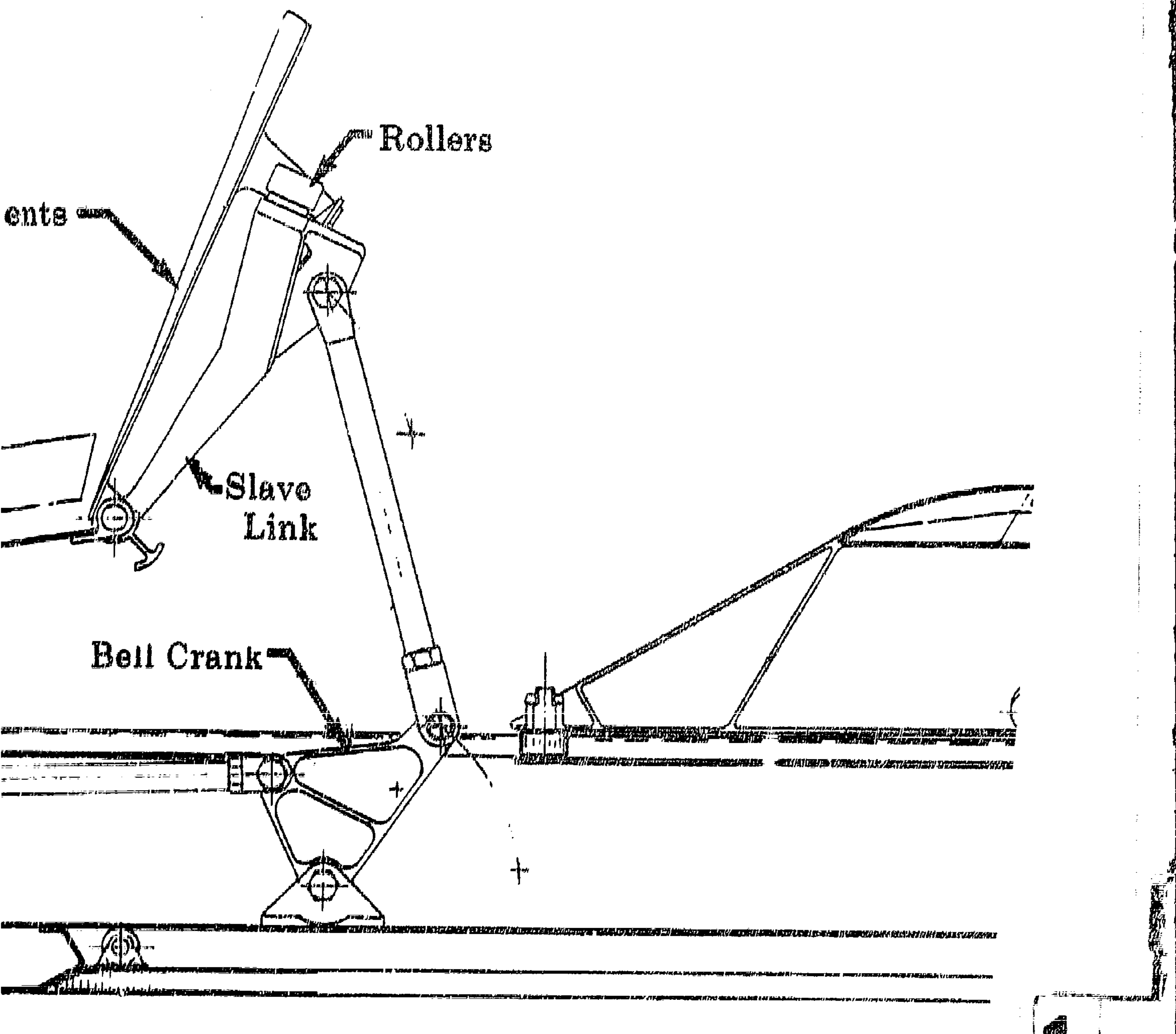
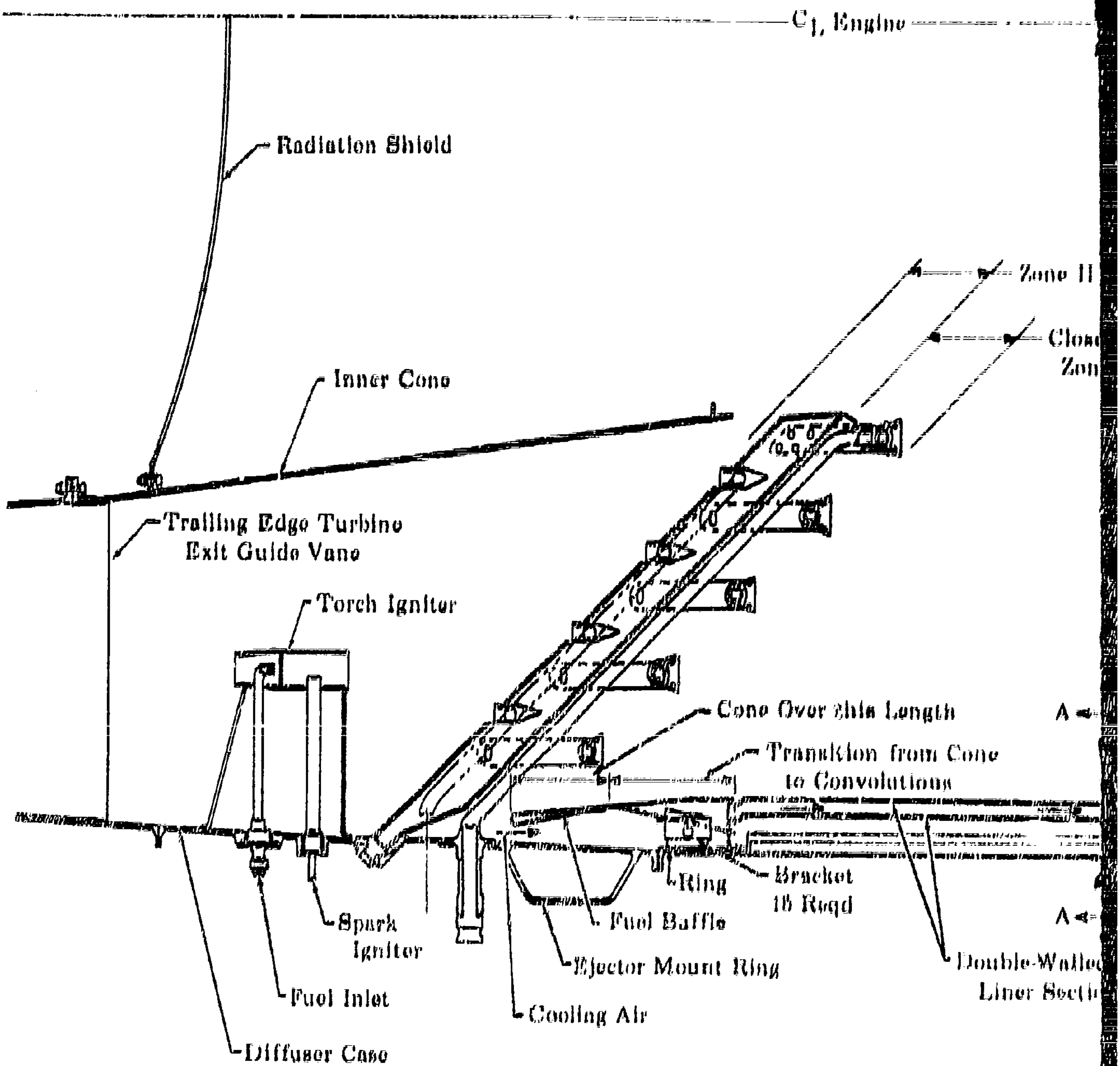
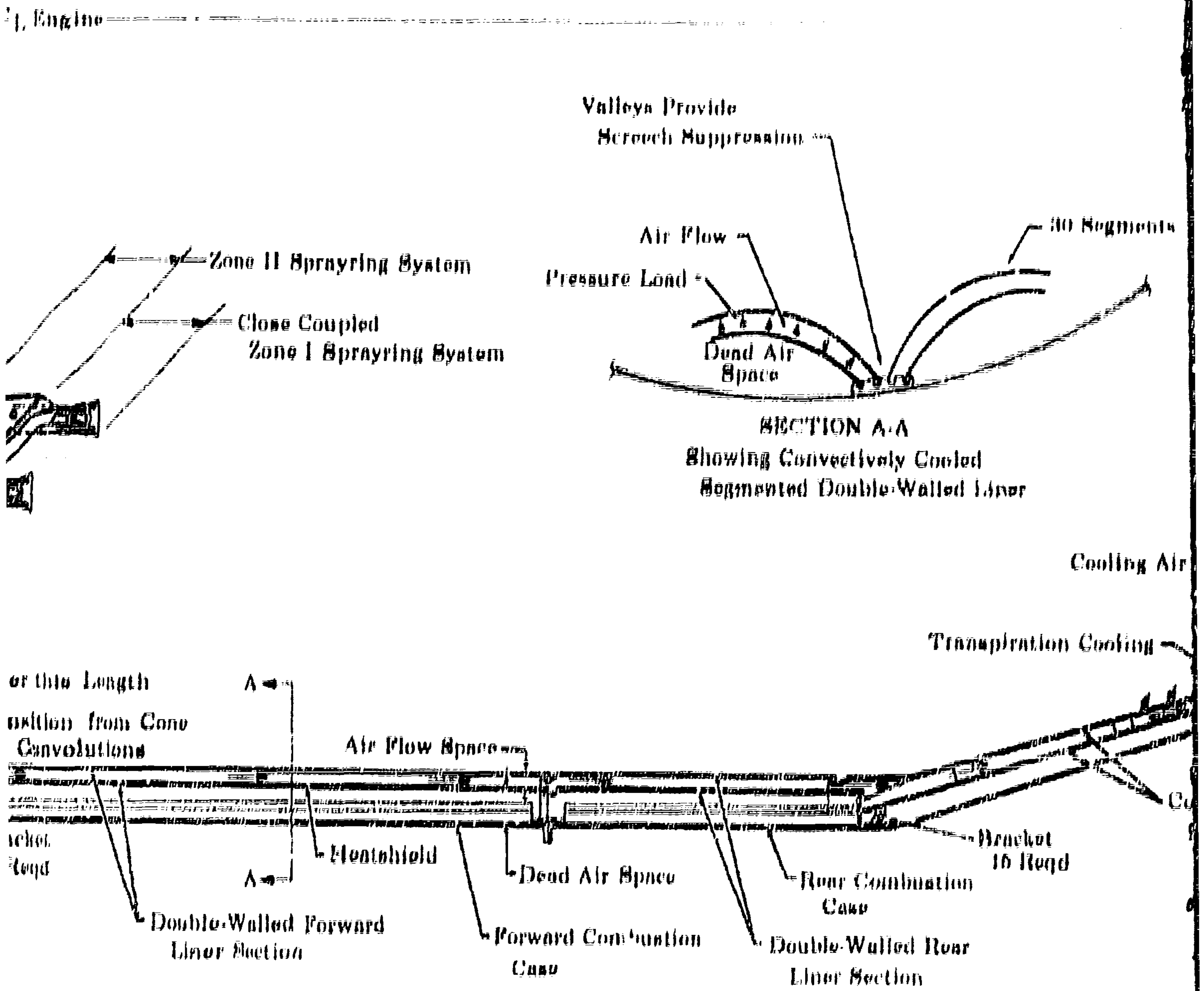



Figure 2B-61

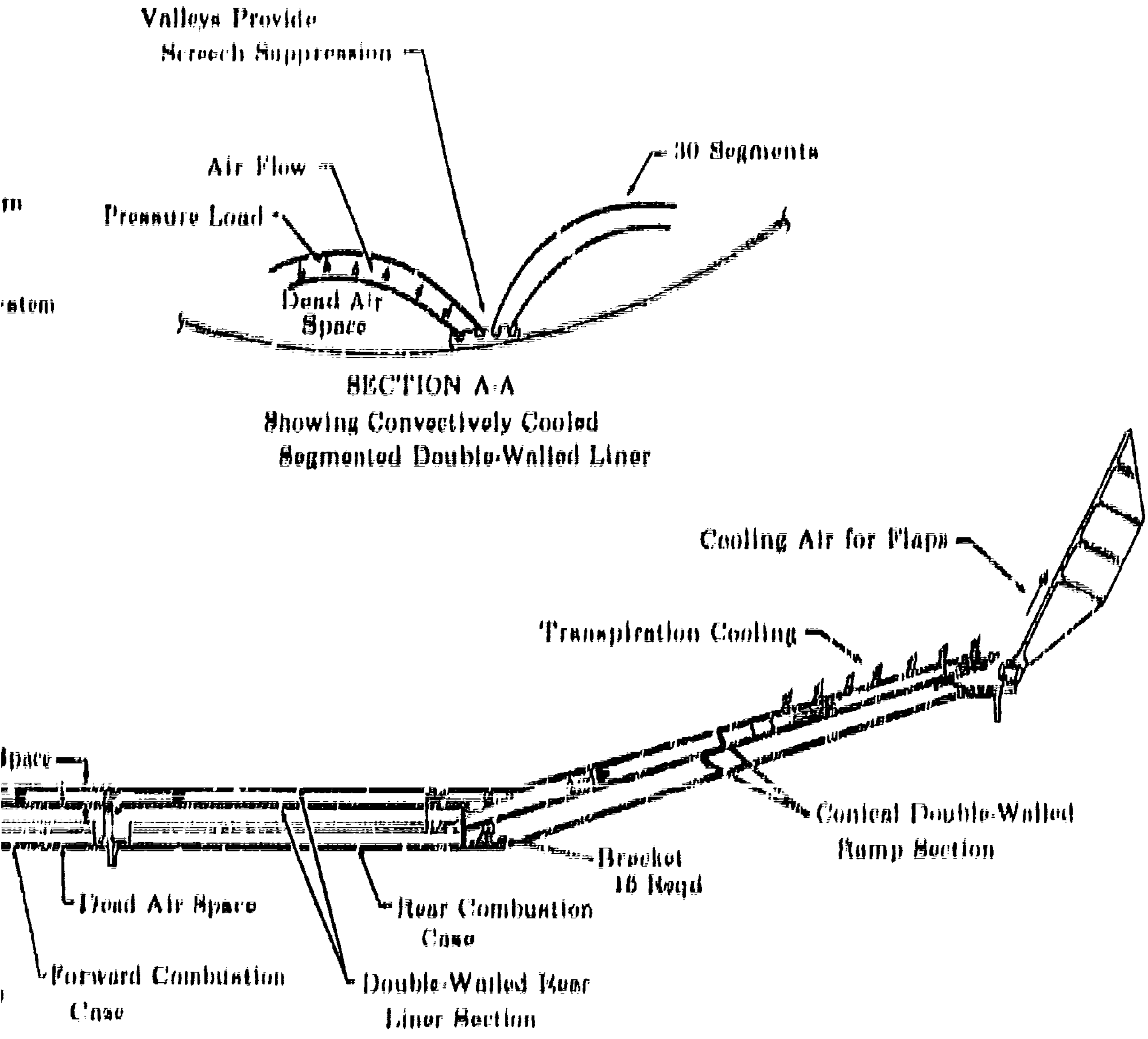


℄, Engine
Radiation Shield
Zone II
Close
Zon
Inner Cone
Trailing Edge Turbine
Exit Guide Vane
Torch Igniter
Cone Over this Length
A
Transition from Cone
to Convolutions
Ring
Bracket
16 Reqd
A
Spark
Igniter
Fuel Baffle
Ejector Mount Ring
Double-Walled
Liner Secti
Fuel Inlet
Cooling Air
Diffuser Case
(Outer)



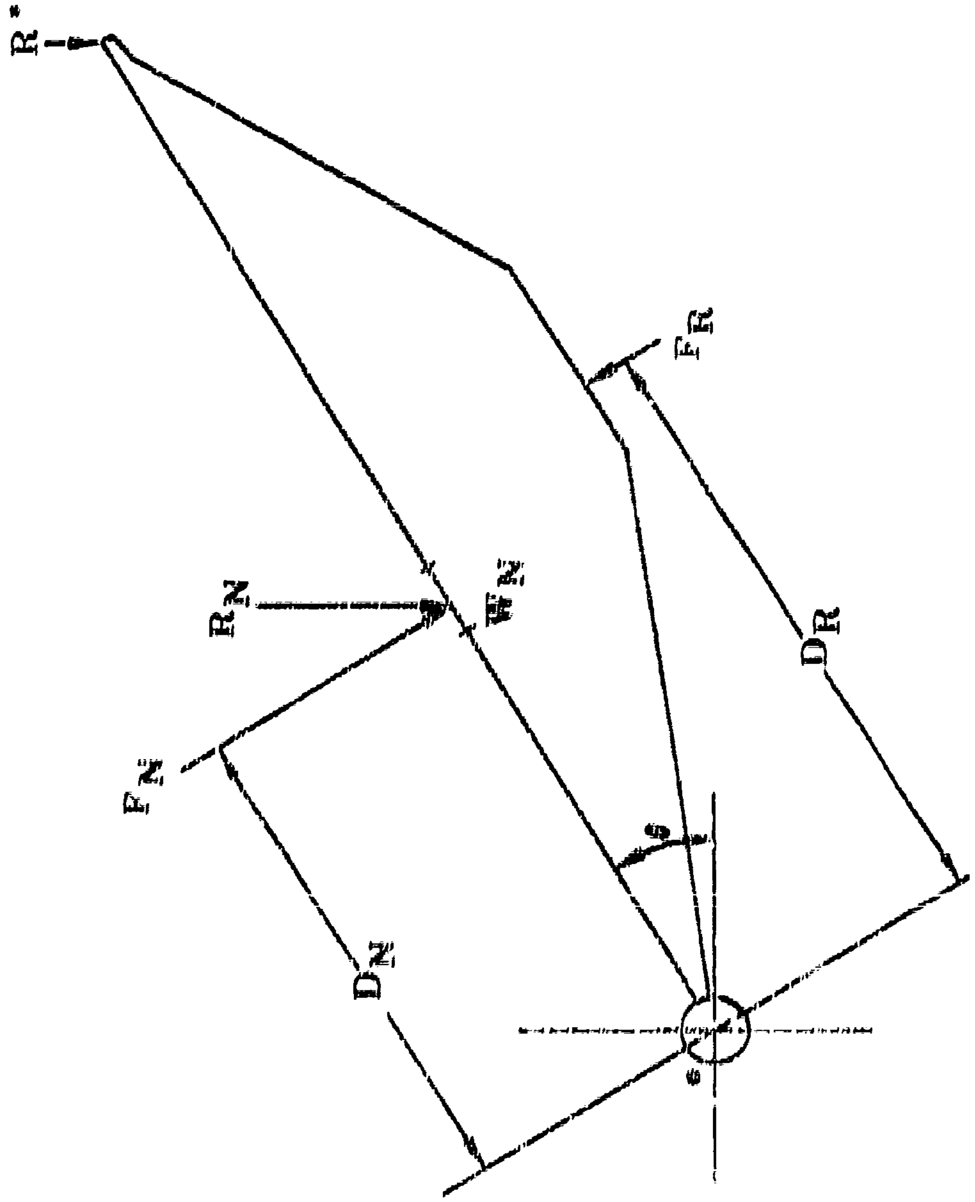

STJ227 AFTERBURNER SYS

Figure 2B-62



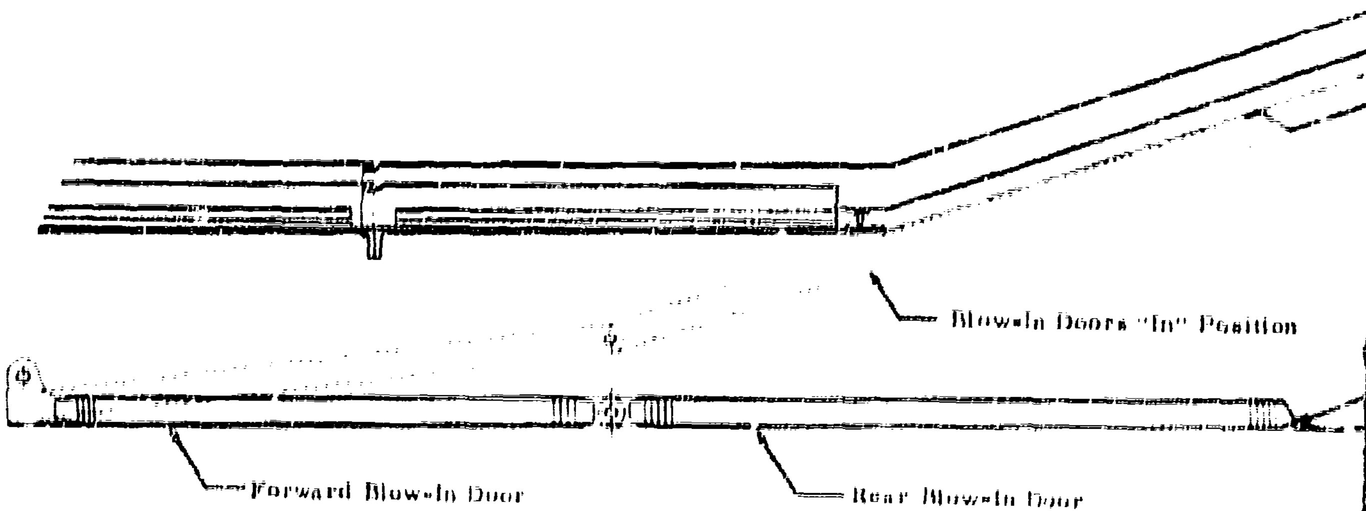

STJ227 AFTERBURNER SYSTEM

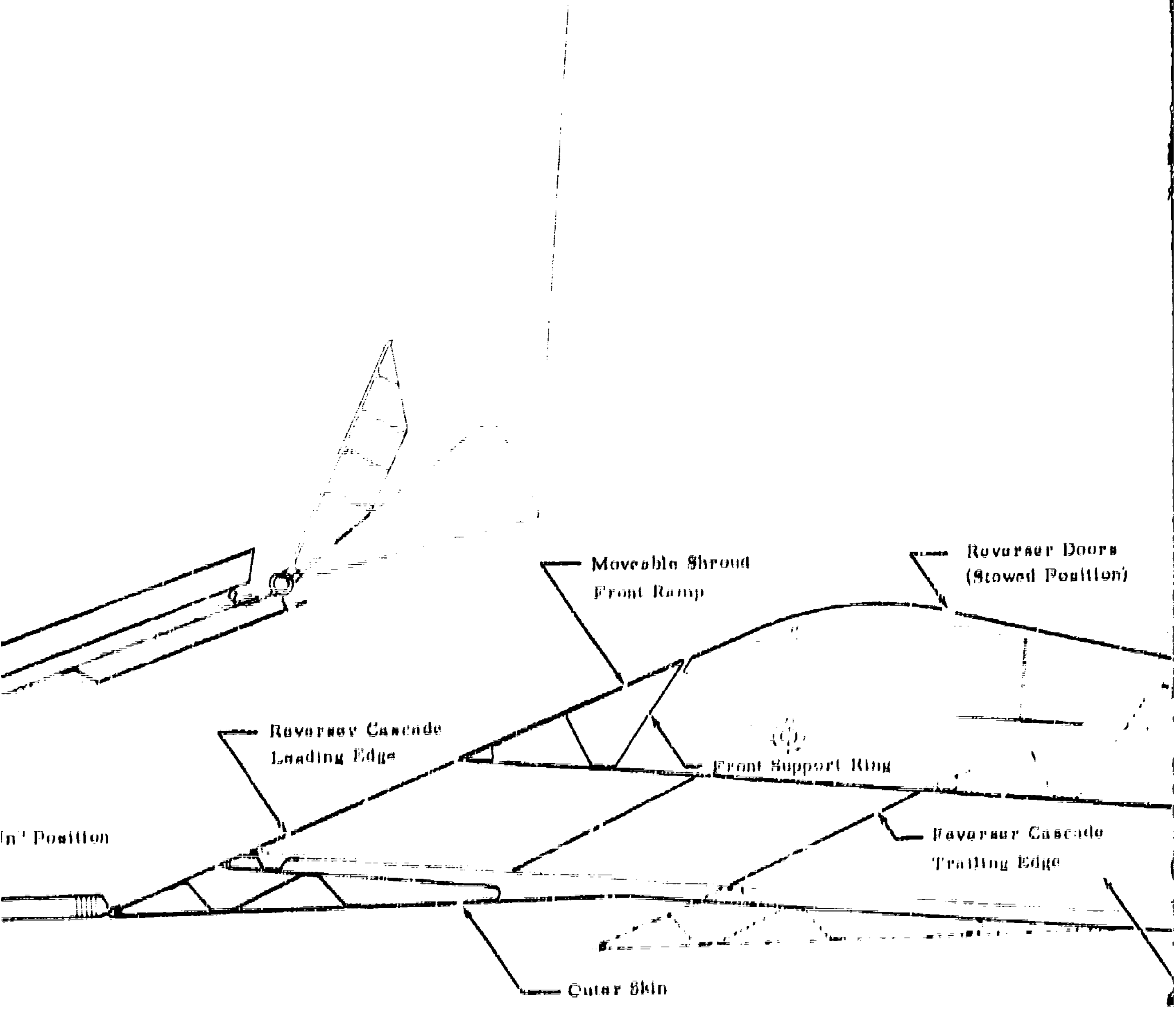
Figure 2B-62



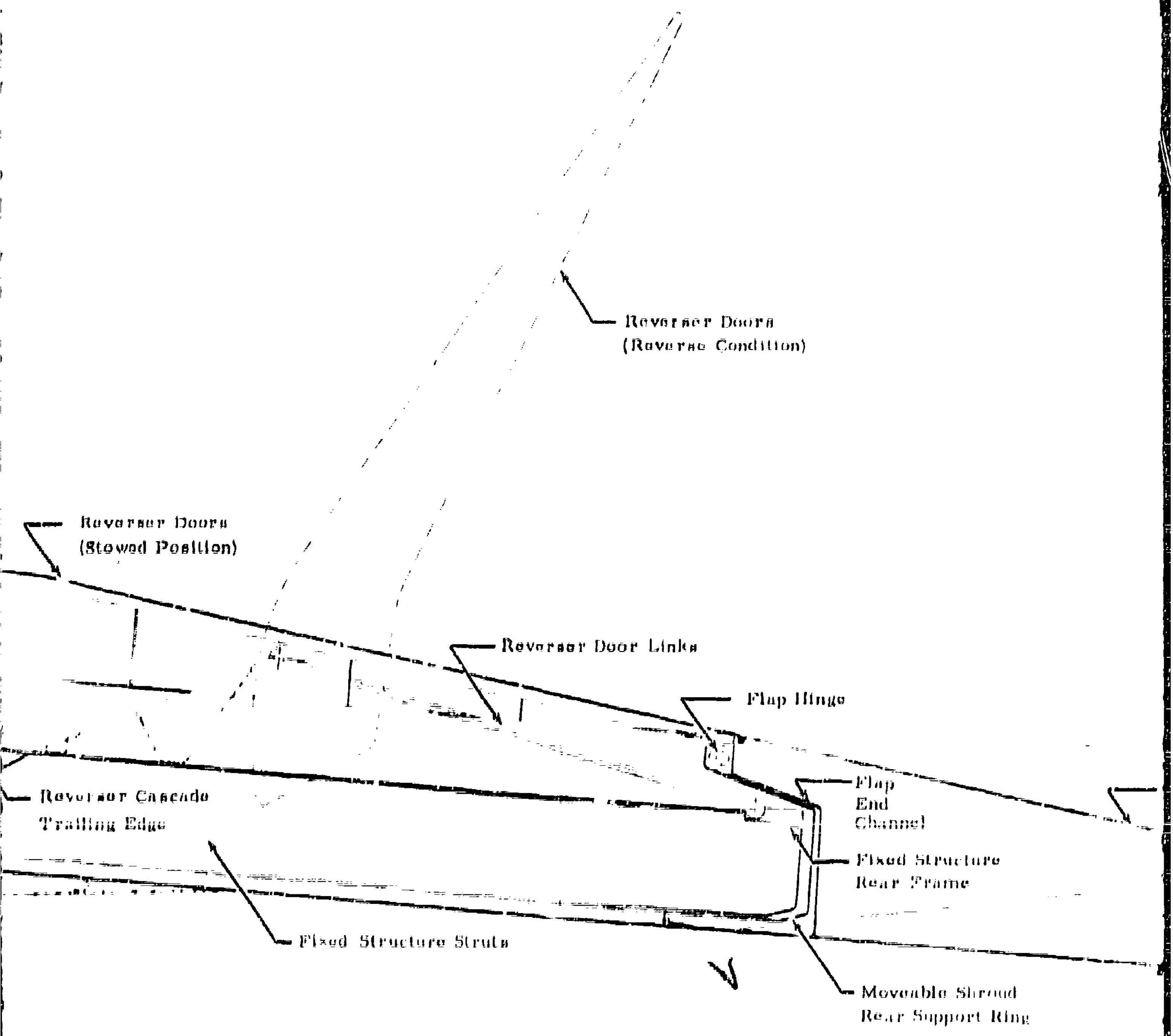
FLAP LOAD CALCULATION

Figure 2B-63

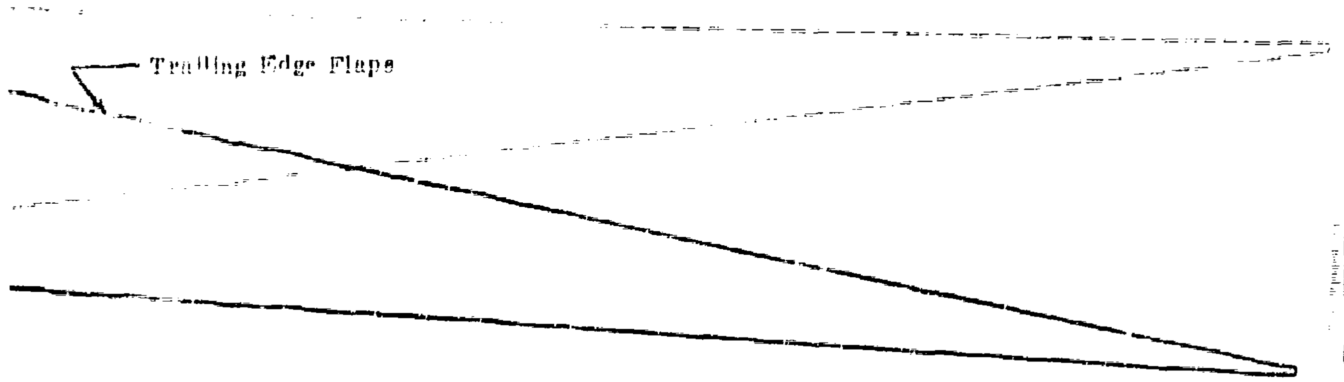

Blow-In Doors "In" Position
Forward Blow-In Door
Rear Blow-In Door


Moveable Shroud
Front Ramp
Reverser Doors
(Stowed Position)
Reverser Cascade
Leading Edge
Front Support Ring
In' Position
Reverser Cascade
Trailing Edge
Outer Skin

2

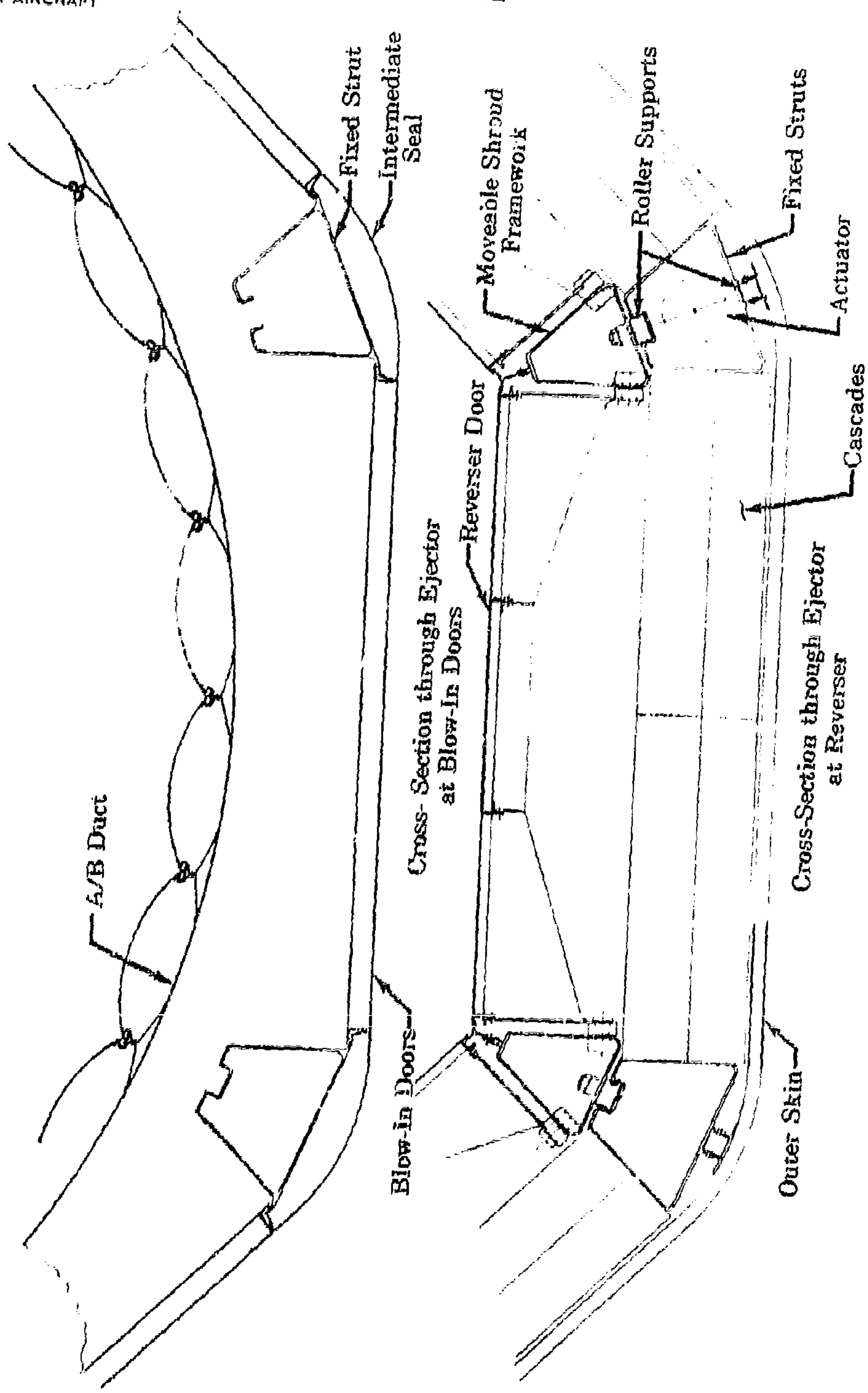

Reverser Doors
(Reverse Condition)
Reverser Doors
(Stowed Position)
Reverser Door Links
Flap Hinge
Flap
End
Channel
Reverser Cascade
Trailing Edge
Fixed Structure
Rear Frame
Fixed Structure Struts
Moveable Shroud
Rear Support Ring

3



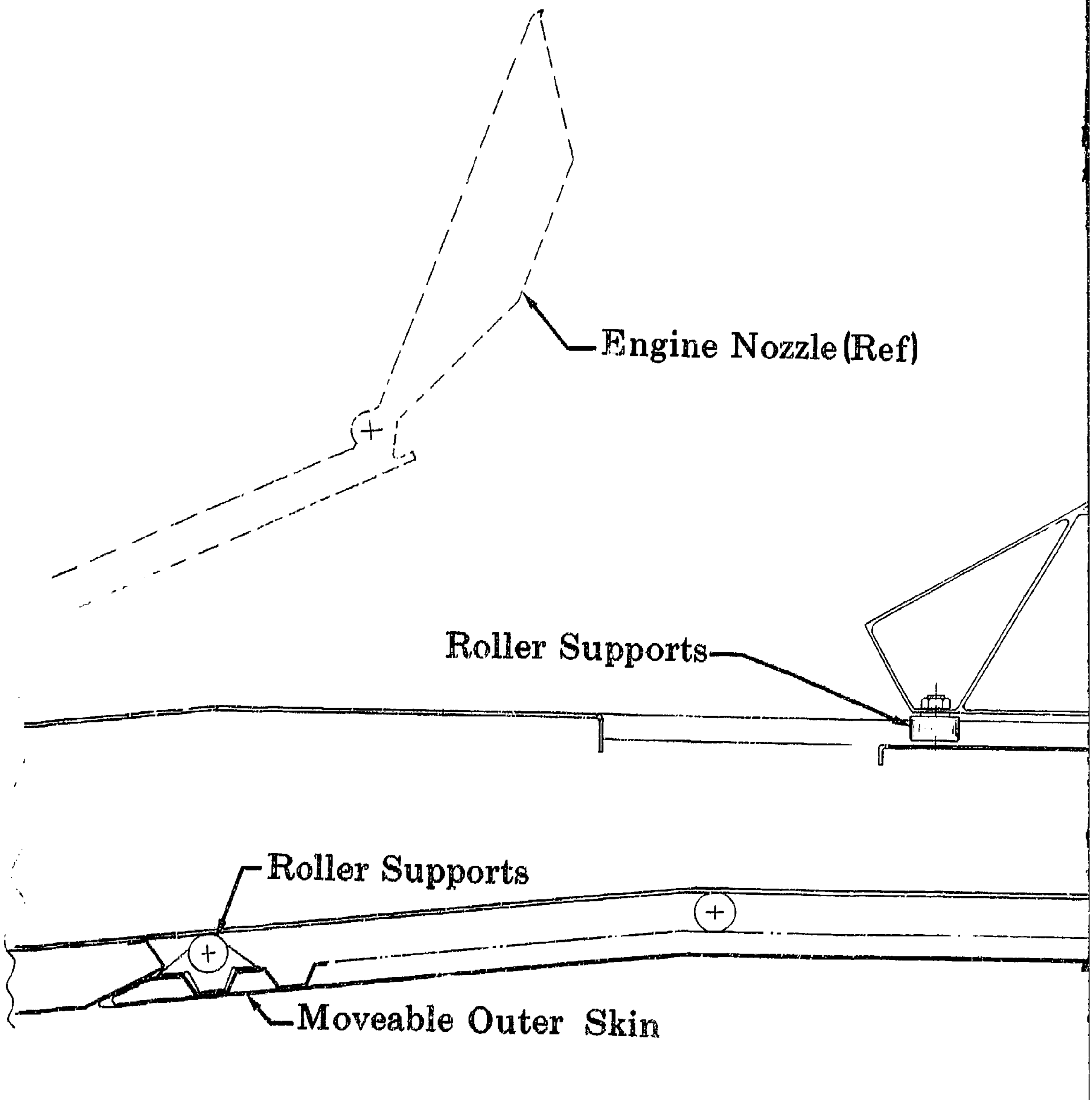

OCTAGONAL BLOW-IN DOOR EJECTOR-REVERSER

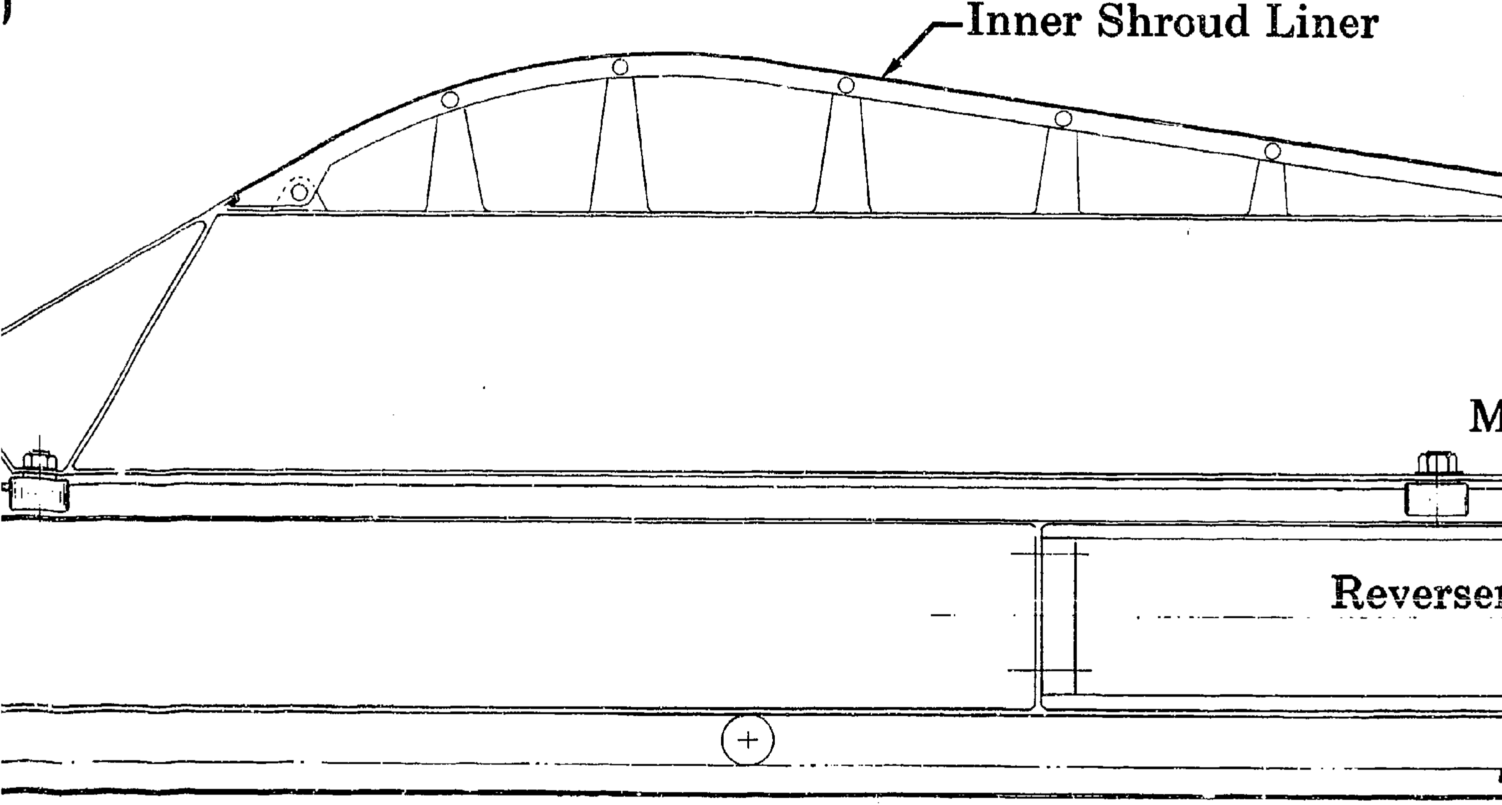
Figure 2B-64



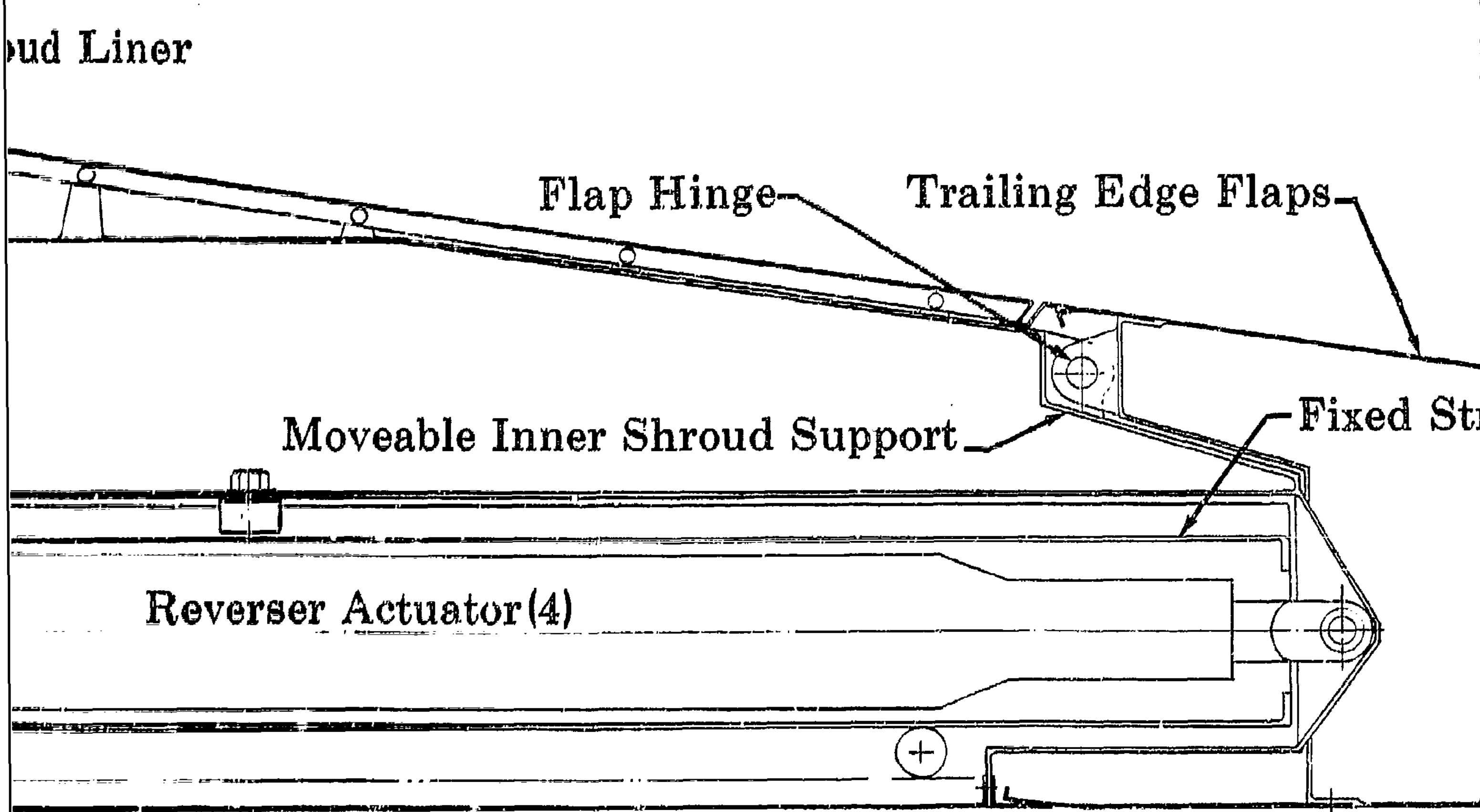

CROSS SECTION THROUGH REVERSER

Figure 2B-65


Engine Nozzle (Ref)
Roller Supports
Roller Supports
Moveable Outer Skin

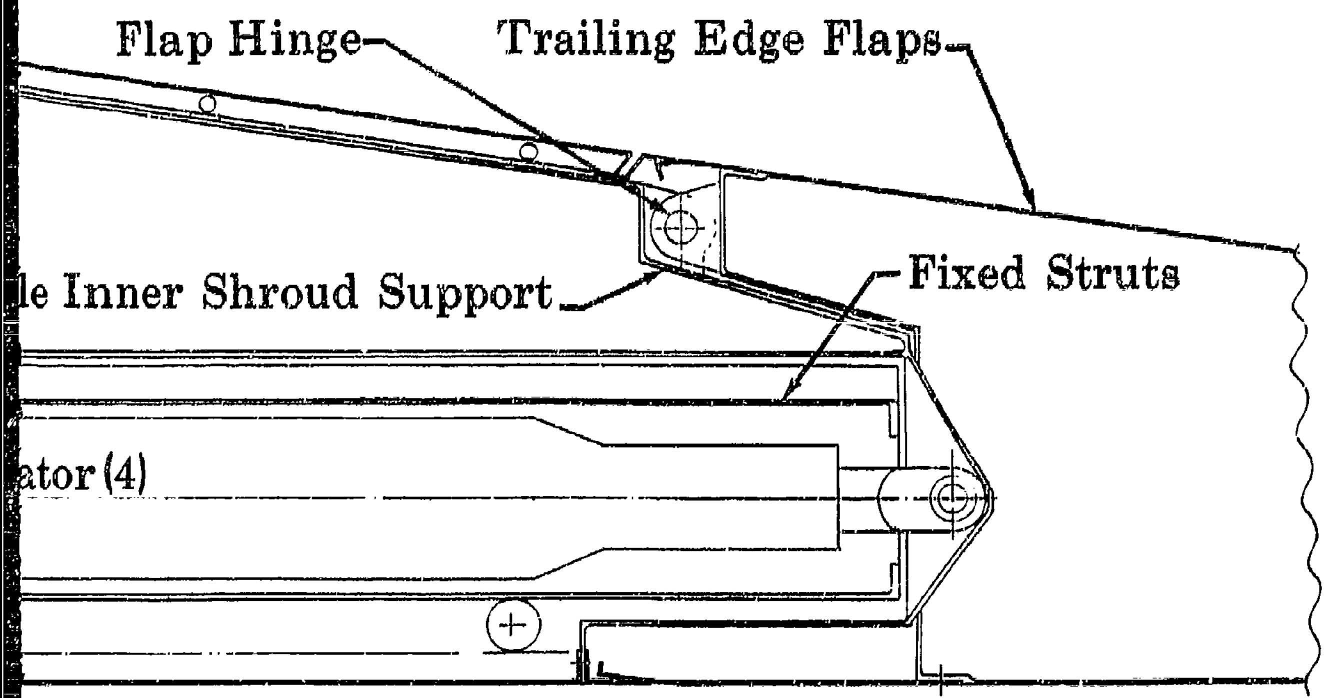



2



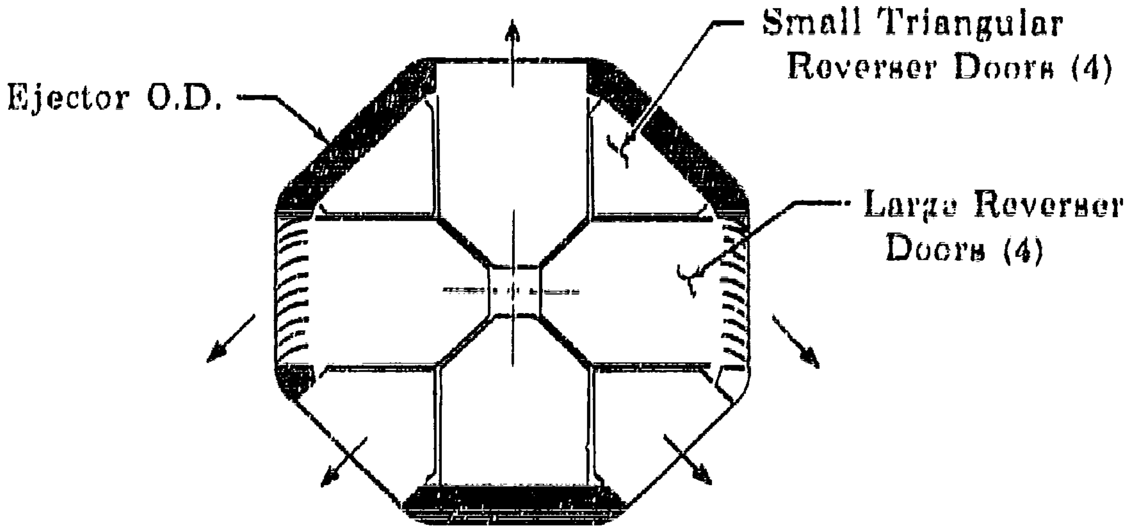

EJECTOR ACTUATION SYSTEM

Figure 2B-66

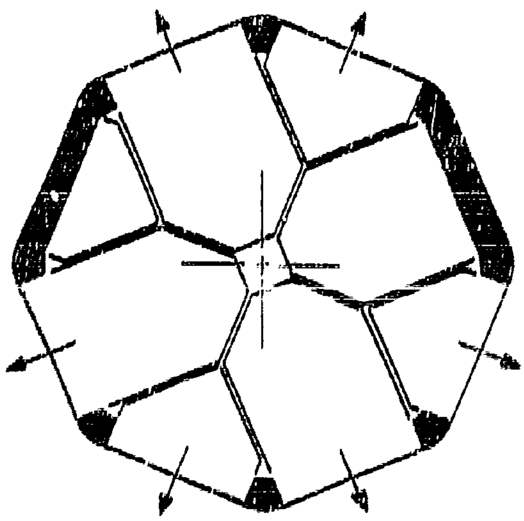



EJECTOR ACTUATION SYSTEM

Figure 2B-66



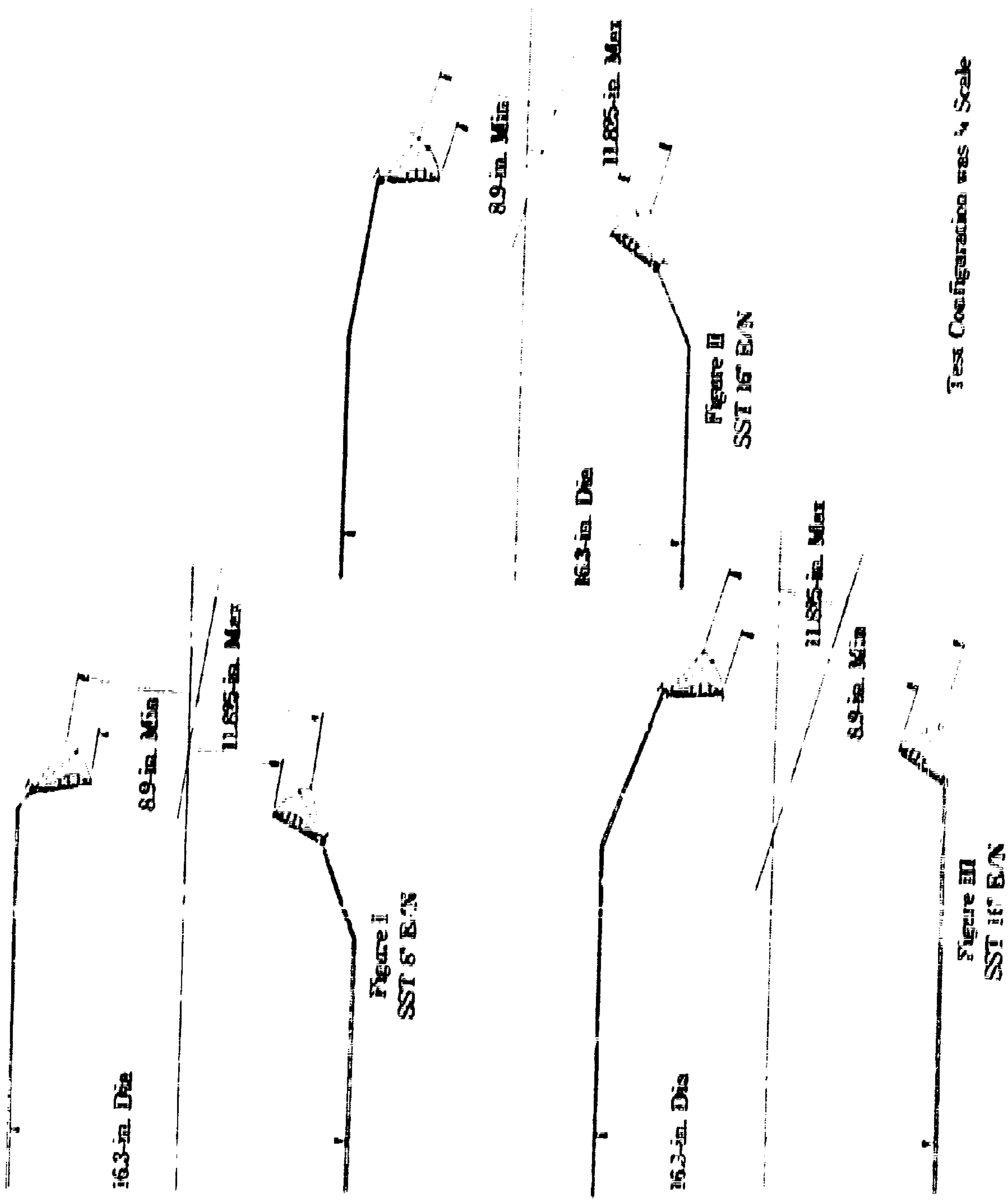

Boeing



Lockheed

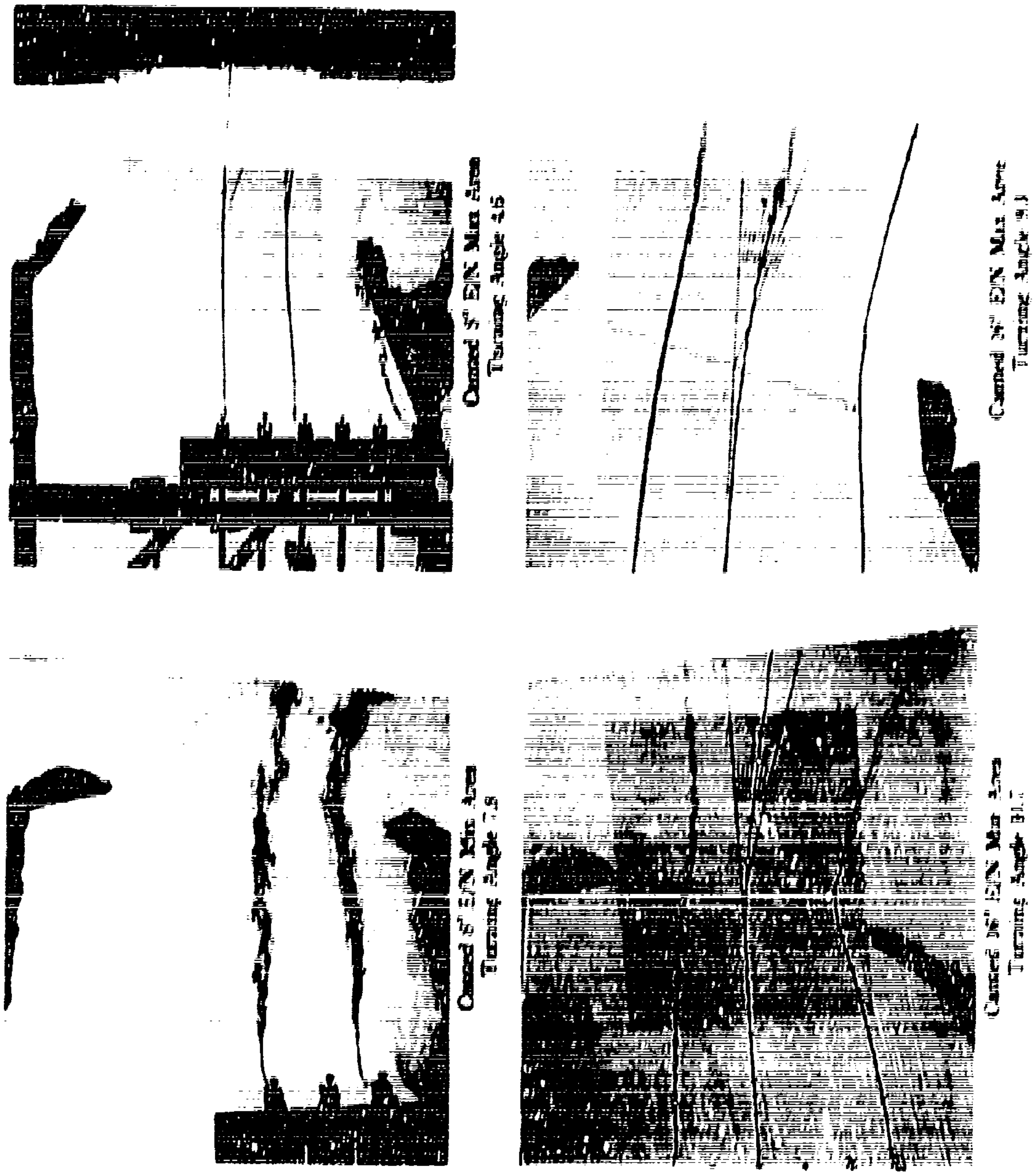
REVERSER TARGETING PATTERNS

Figure 2B-67



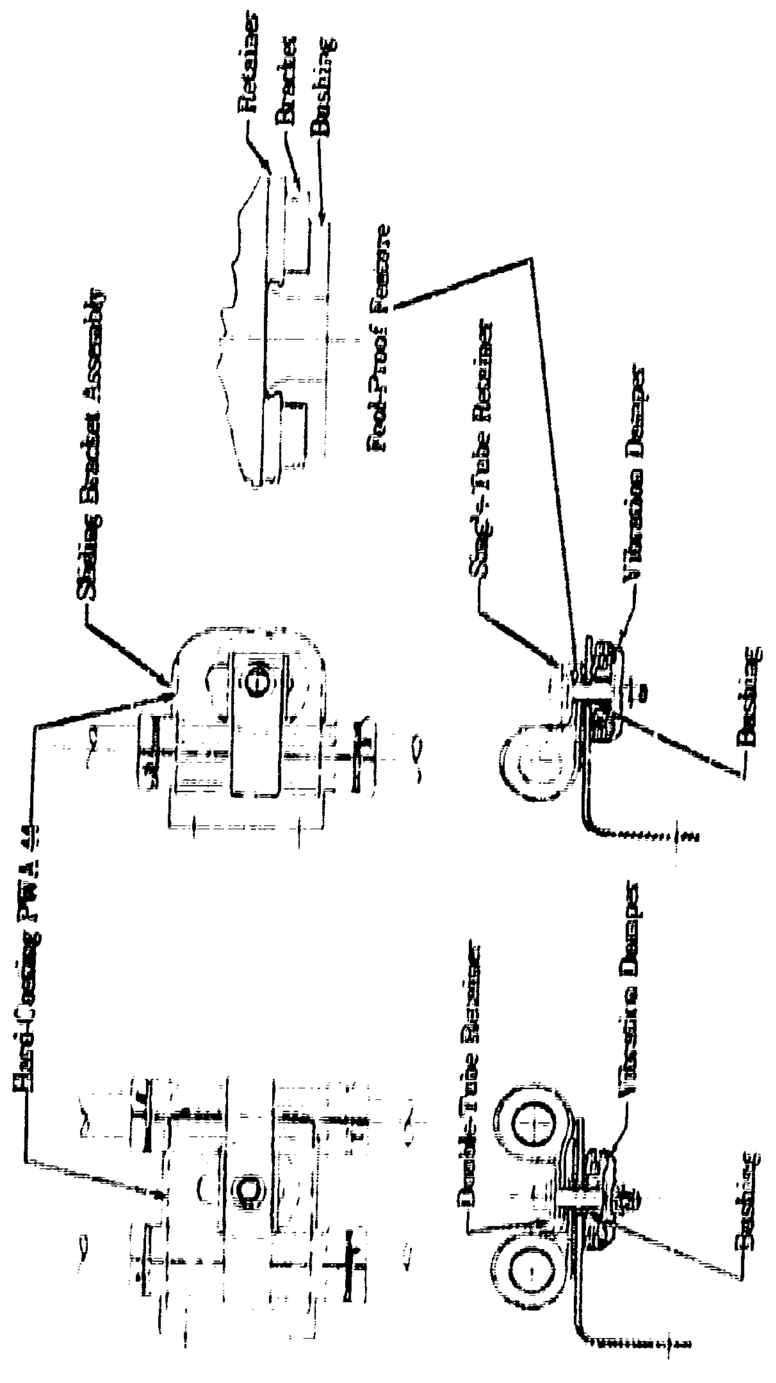

EXHAUST NOZZLE CONFIGURATIONS

Figure 2B-60



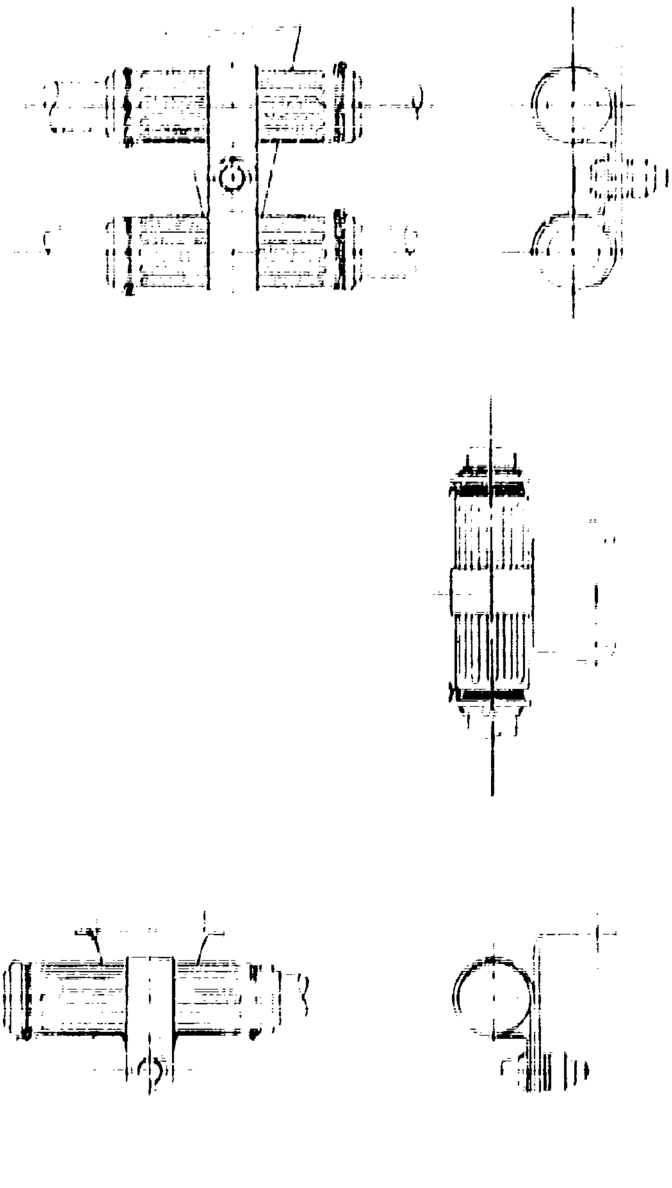
EXHAUST NOZZLE HYDRAULIC ANALOGY

Figure 2B-69



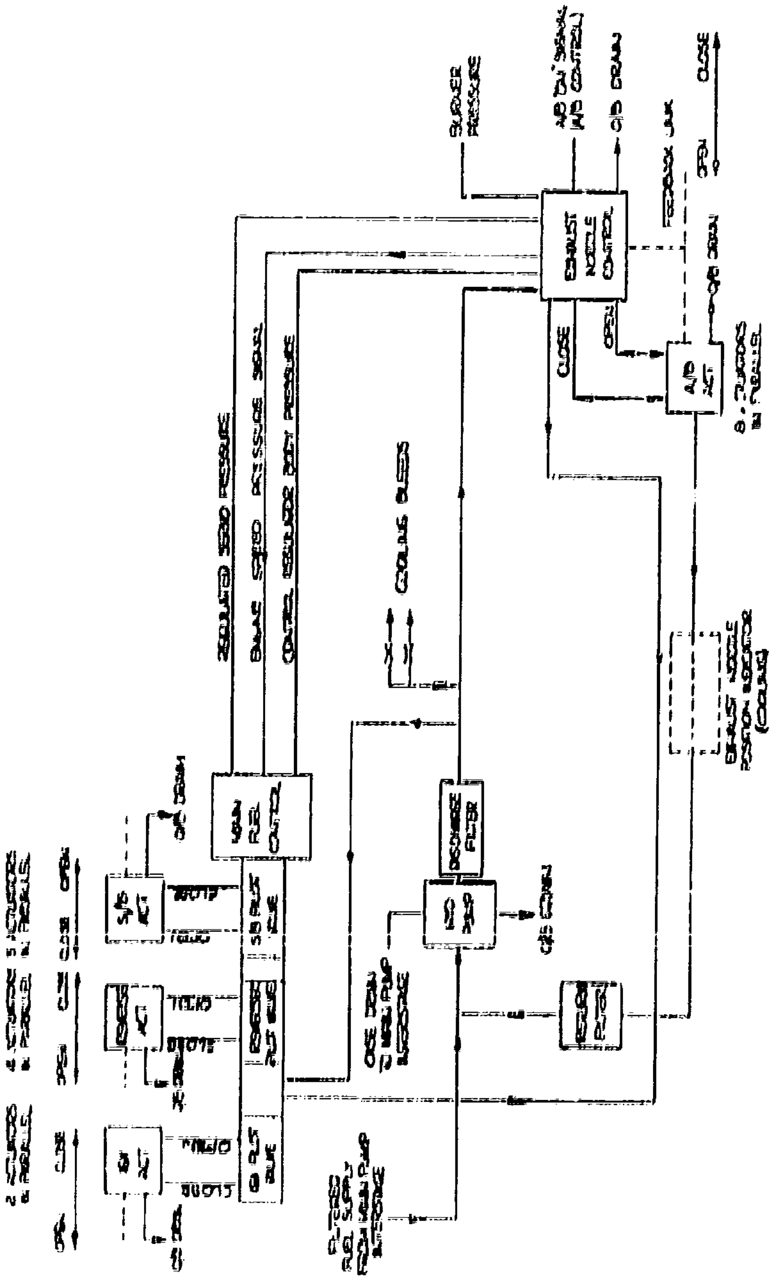

SLIDING BRACKET ASSEMBLY

Figure 2B-70



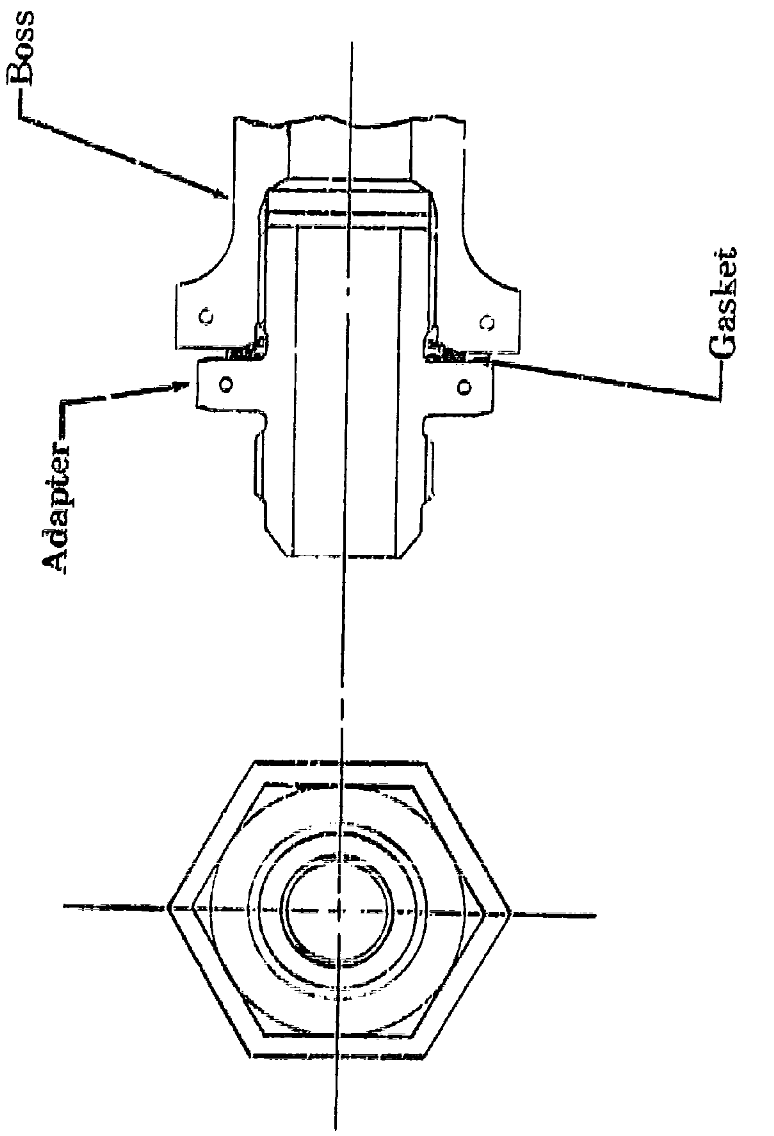
FIXED BRACKET ASSEMBLY

Figure 2B-71



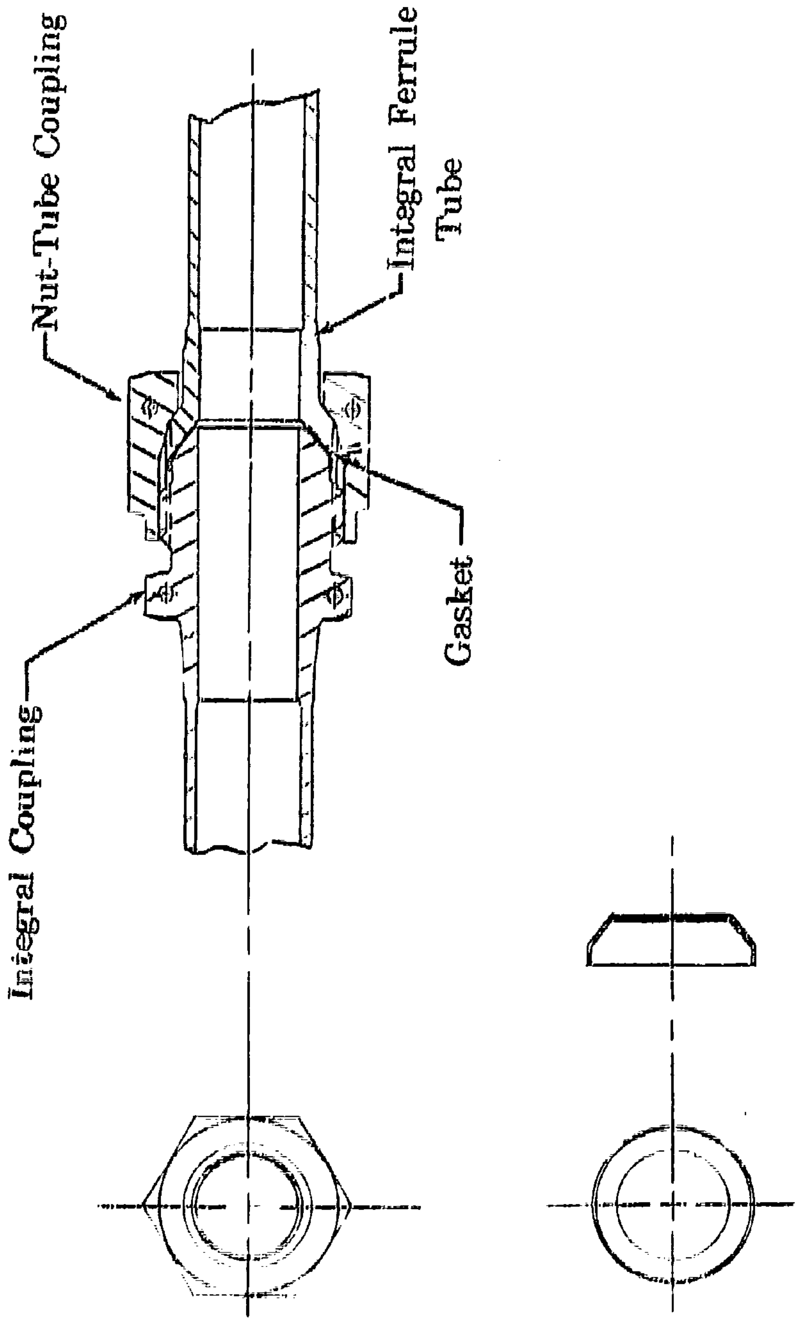
HYDRAULIC SYSTEM

Figure 2B-72



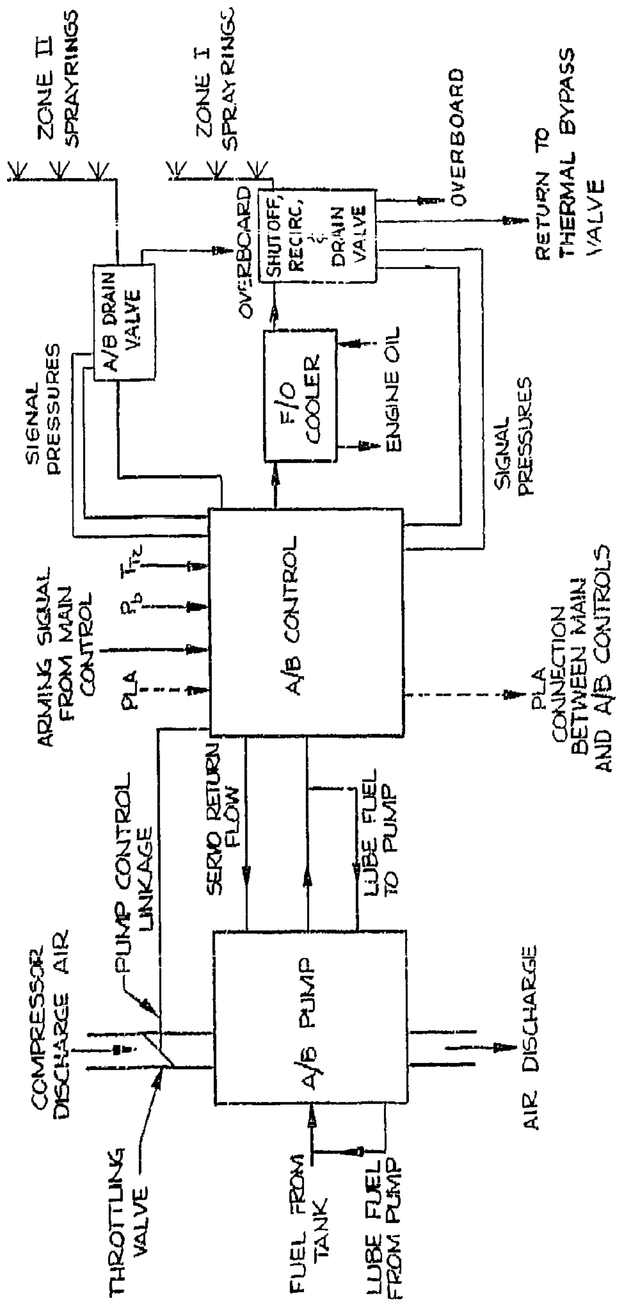

BOSS-TO-ADAPTER JOINT

Figure 2B-73



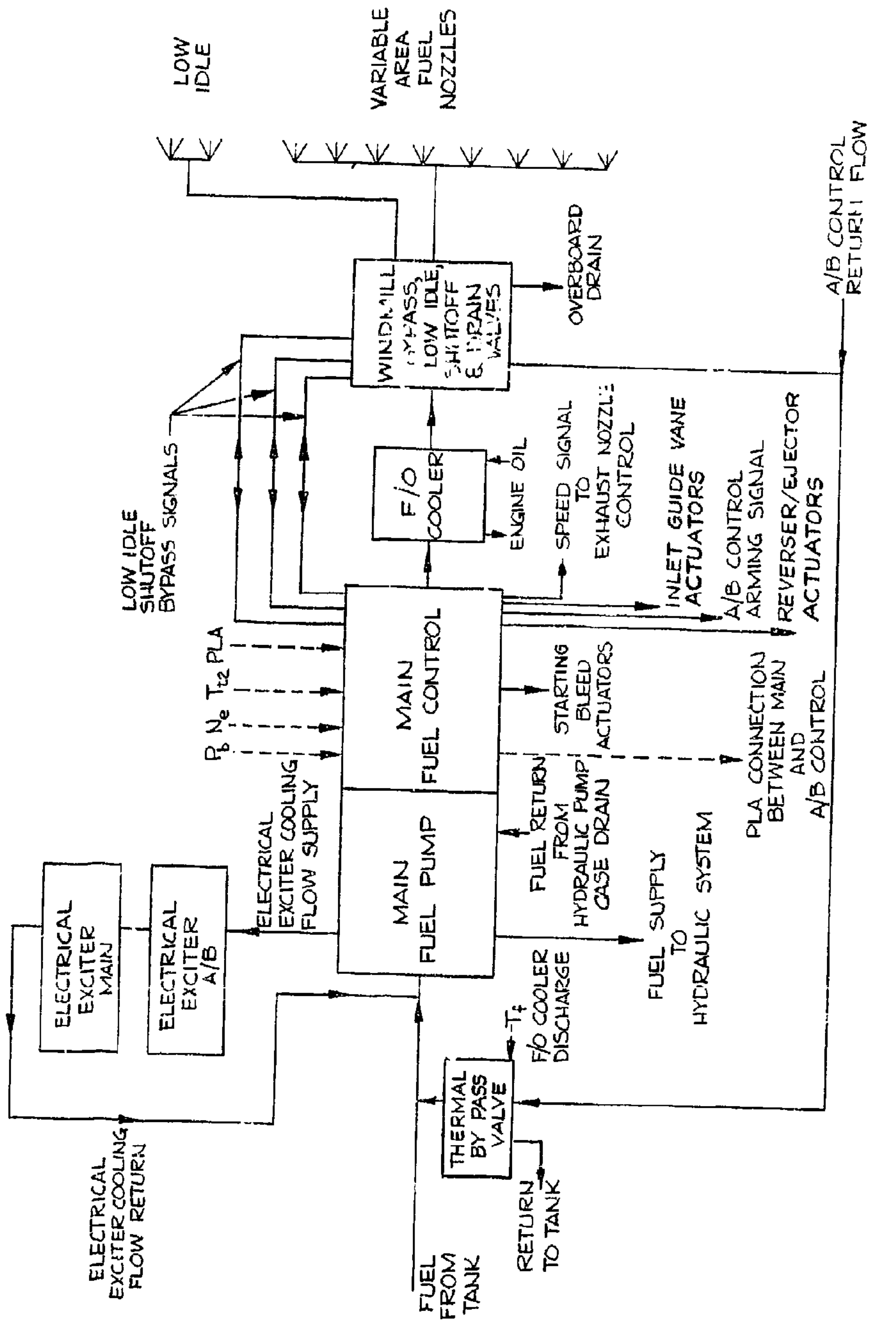

INTEGRAL FERRULE JOINT WITH SEALING GASKET

Figure 2B-74



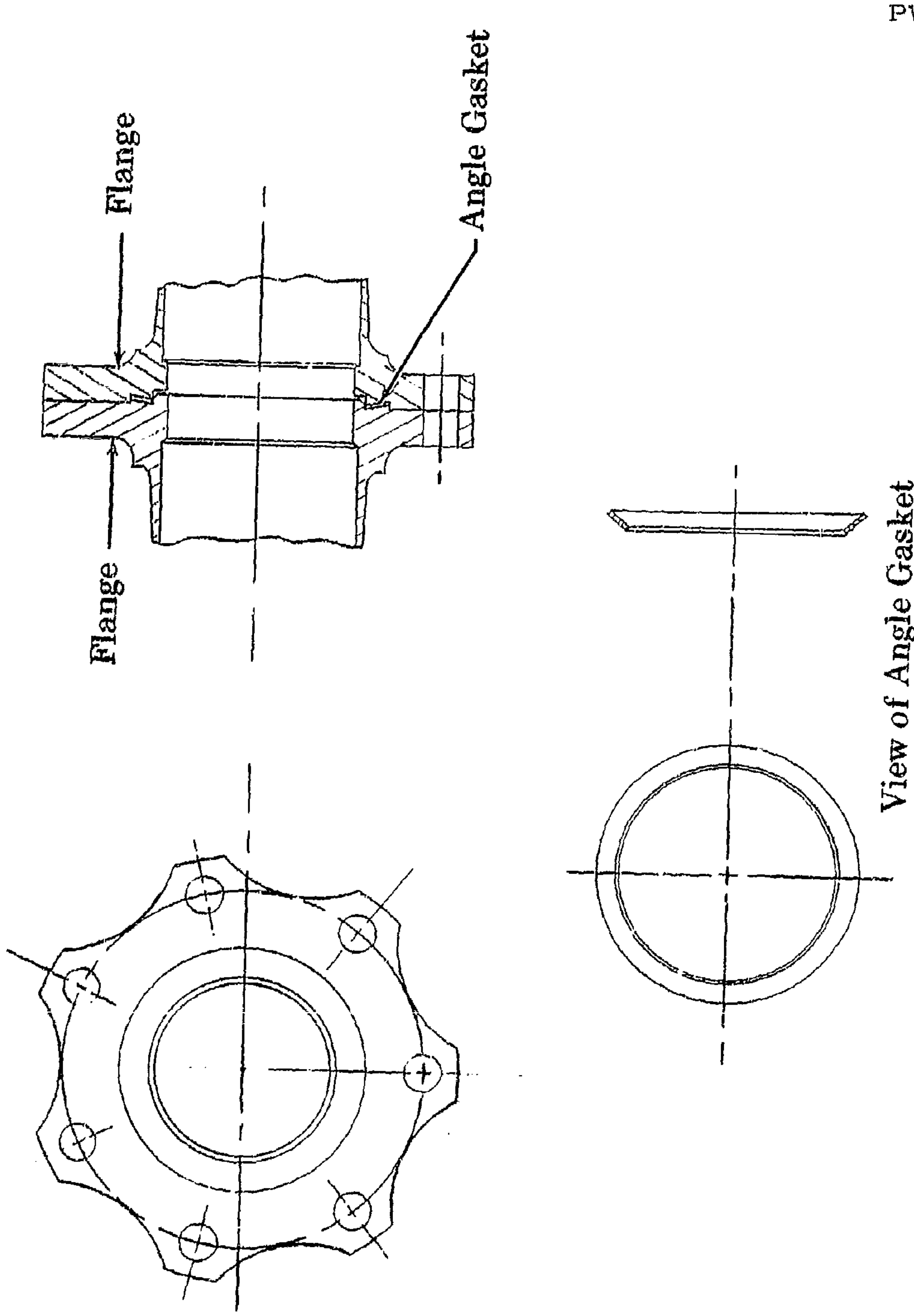

AFTERBURNER FUEL SYSTEM

Figure 2B-75



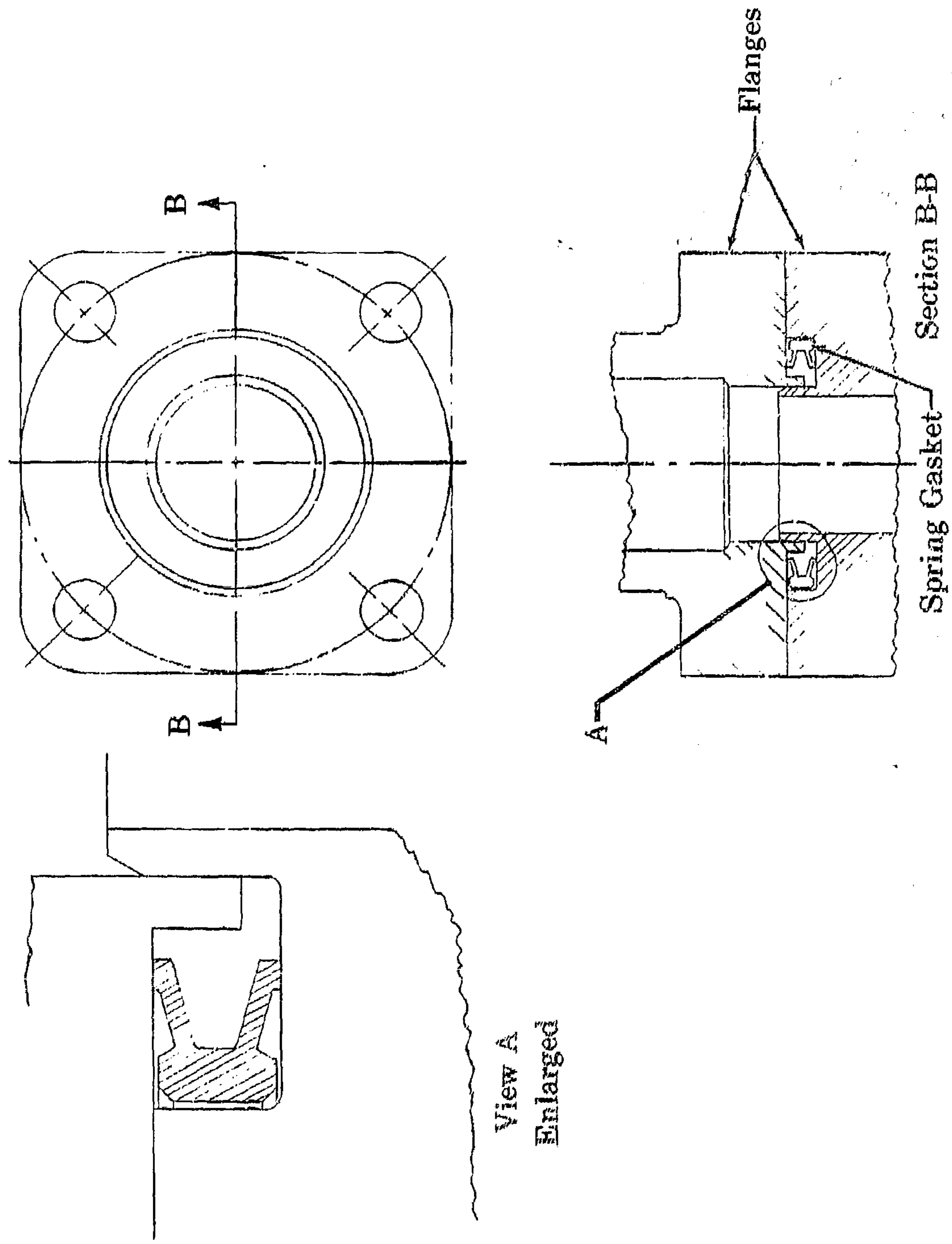

MAIN FUEL SYSTEM

Figure 2B-76



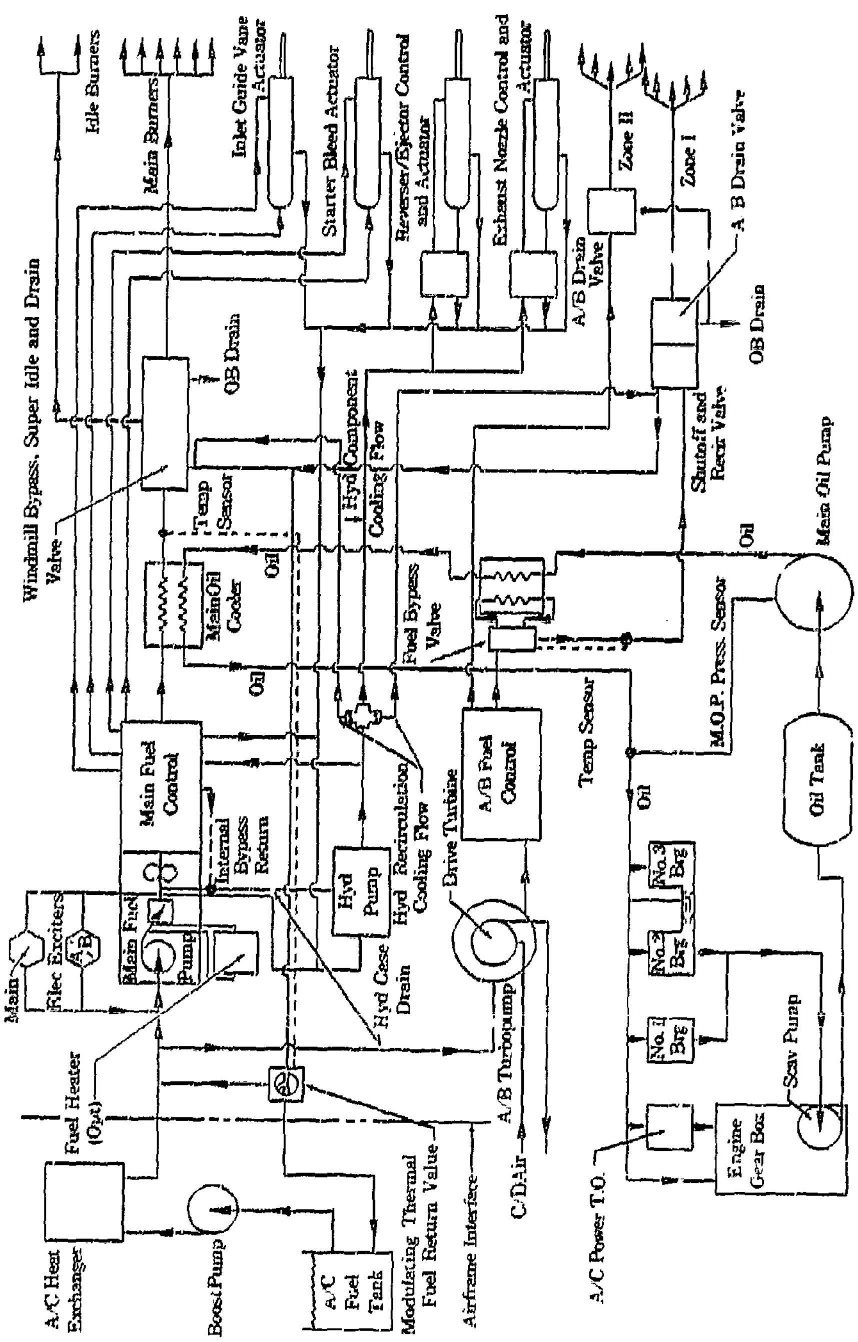

METALLIC STATIC SEAL FOR JOINT SUBJECT
TO MODERATE THERMAL SHOCK

Figure 2B-77



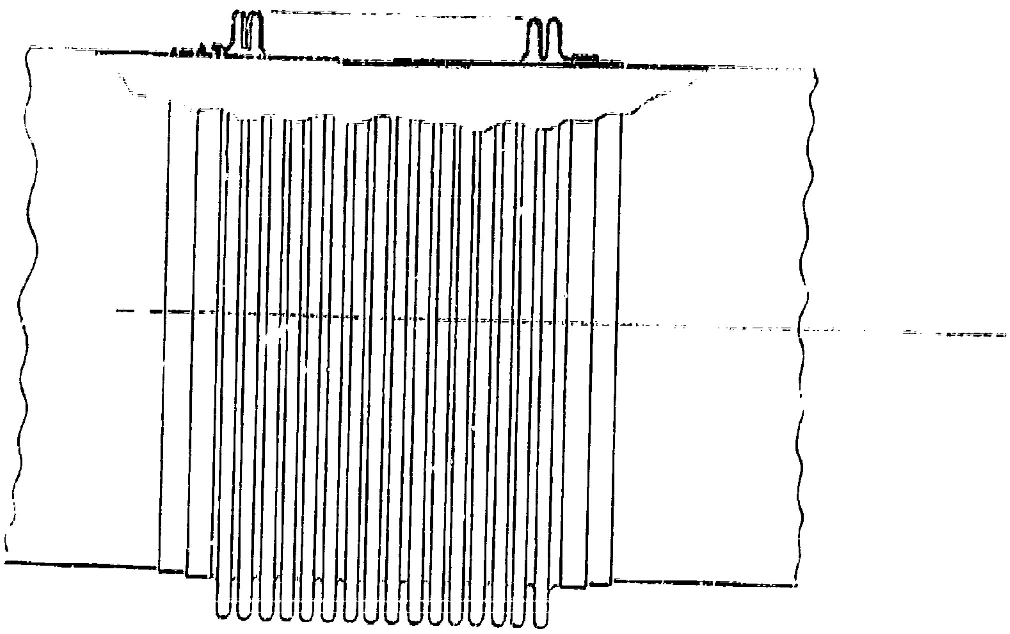

METALLIC STATIC SEAL FOR JOINTS SUBJECT
TO HIGH THERMAL SHOCK

Figure 2B-78



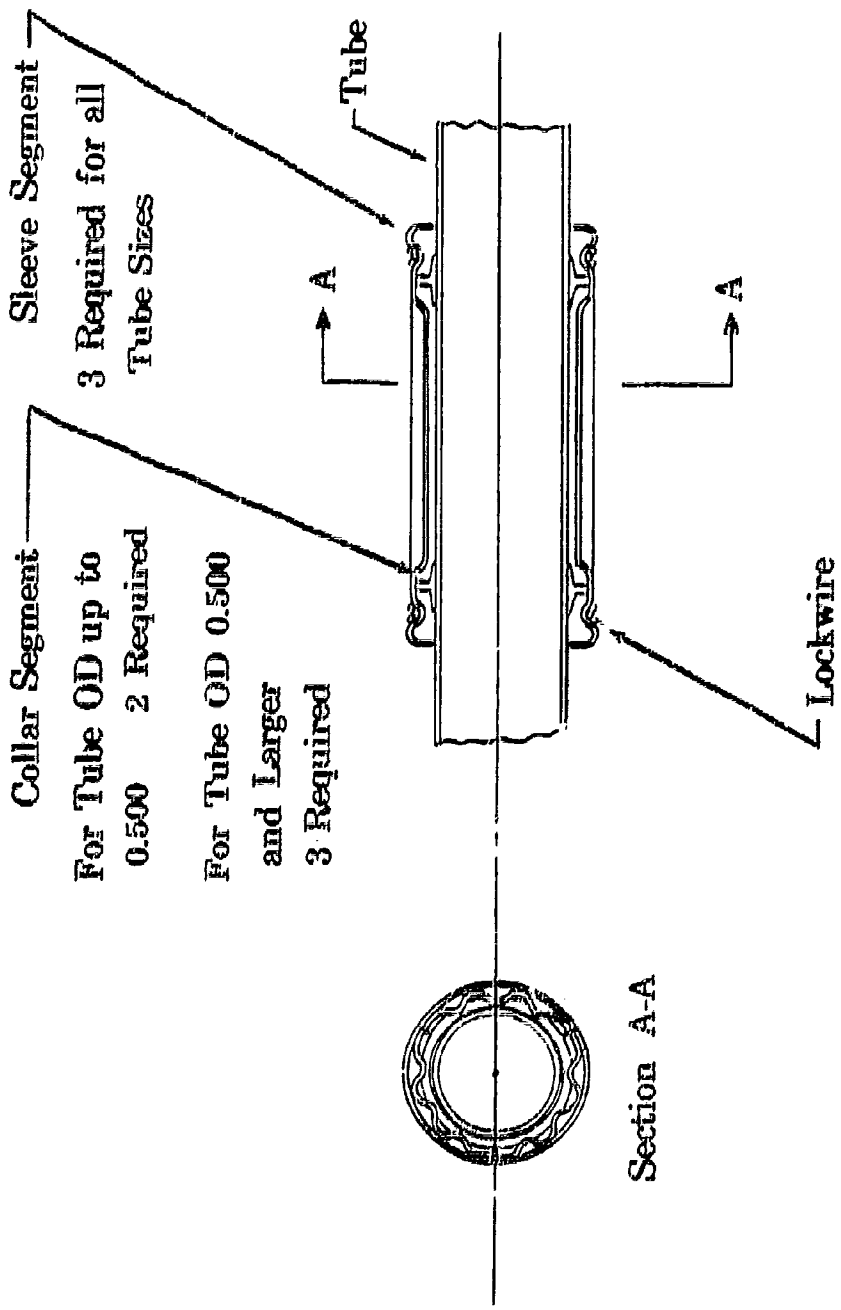

STJ227 FUEL, HYDRAULIC, AND LUBRICATION SYSTEM SCHFMATIC

Figure 2B-79



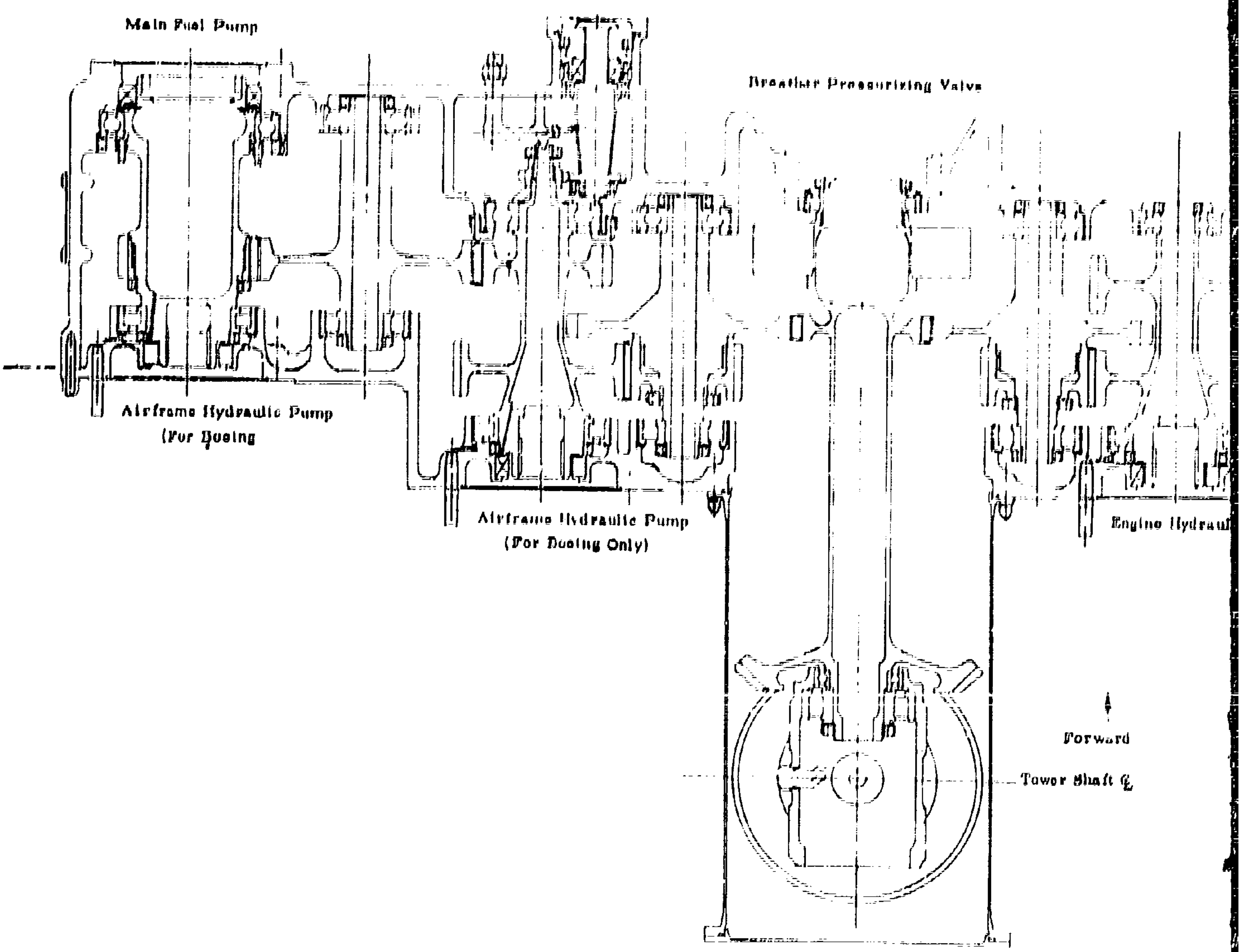
BELLOWS EXPANSION JOINT

Figure 2B-80

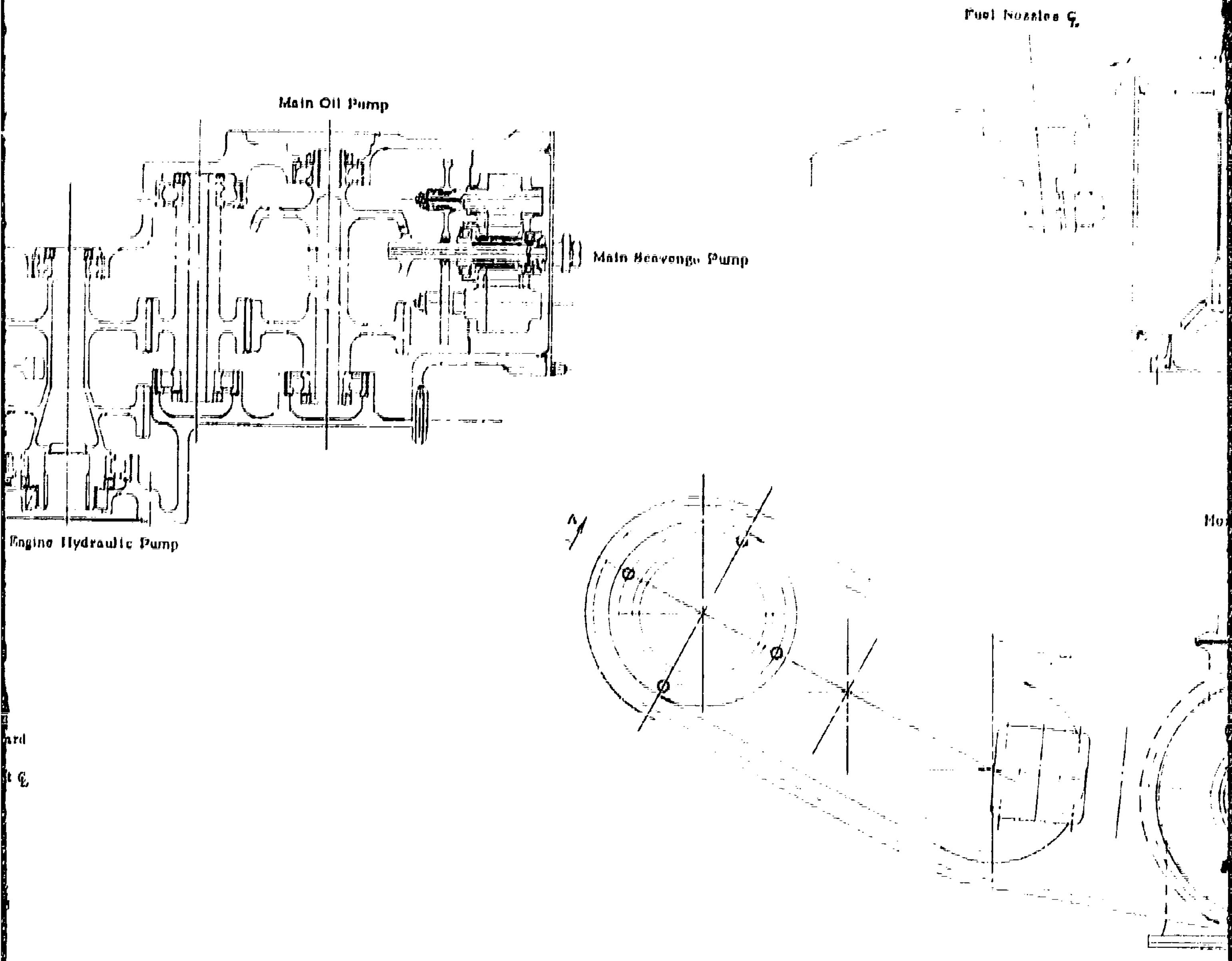



TUBE SUPPORT CLAMP STAND-OFF

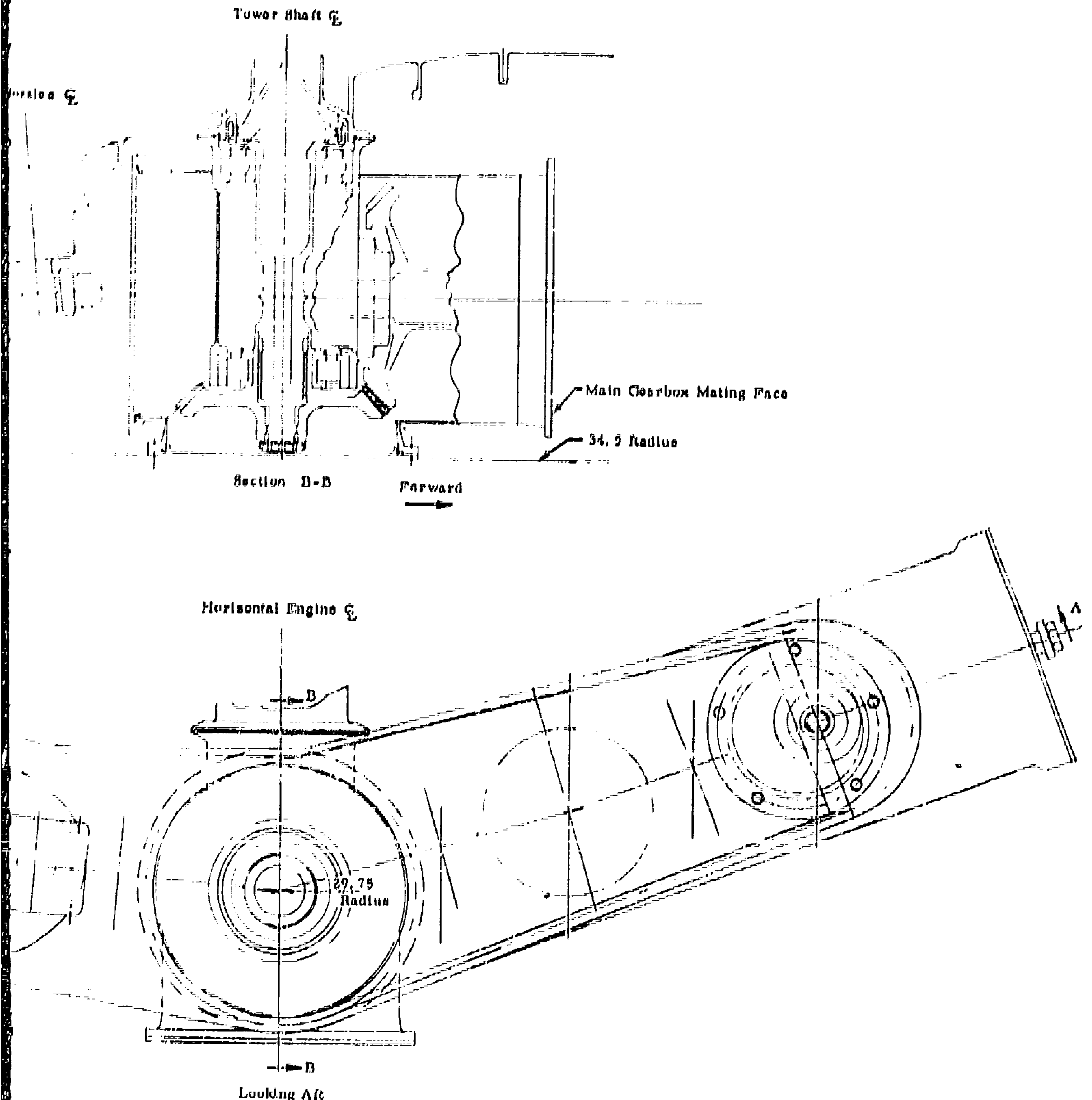
Figure 2B-81


Tachometer
Main Fuel Pump
Breather Pressurizing Valve
Airframe Hydraulic Pump
(For Boeing
Airframe Hydraulic Pump
(For Boeing Only)
Engine Hydraul
Forward
Tower Shaft ℄

Section A-A

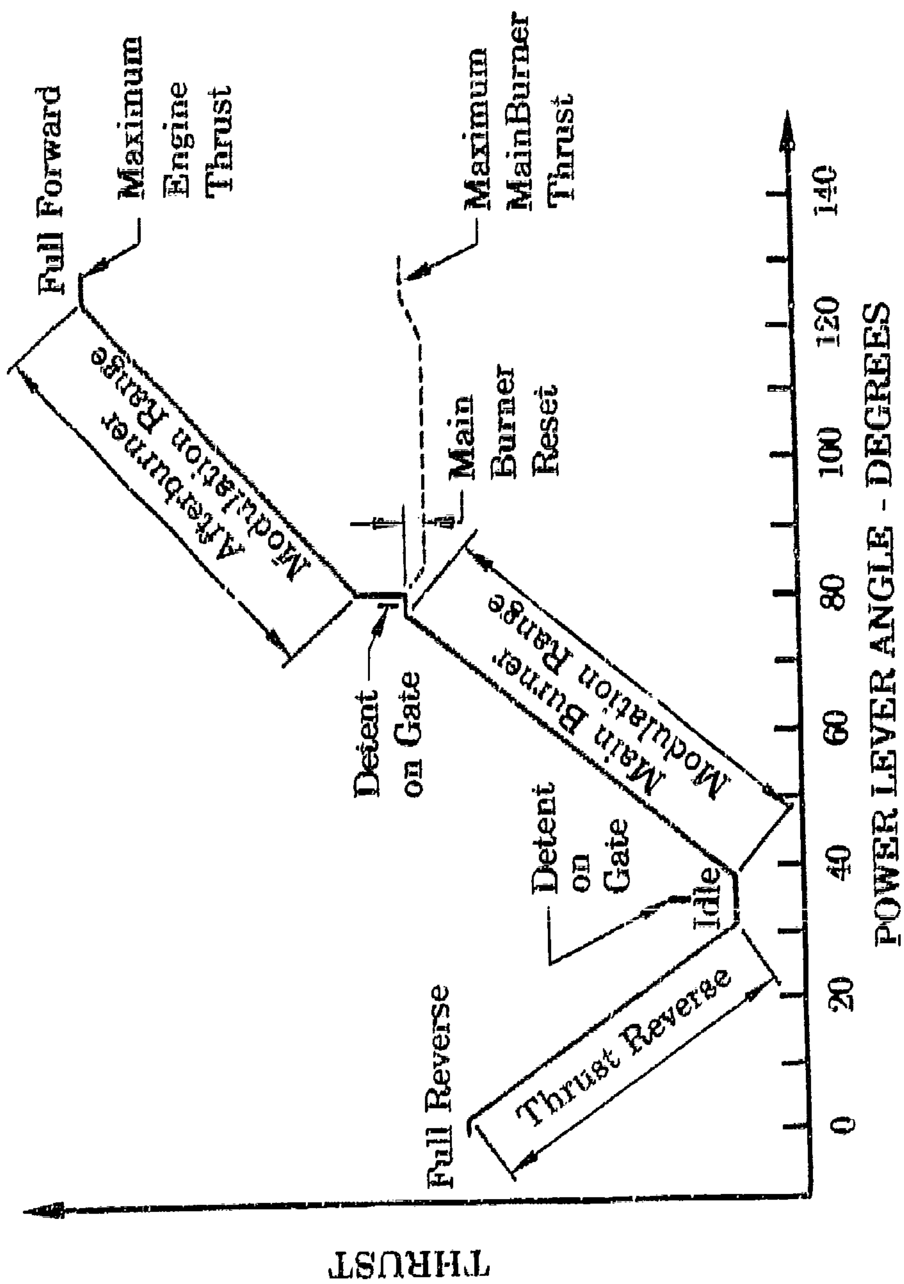

Main Oil Pump
Main Scavenge Pump
Fuel Nozzles ℄
Engine Hydraulic Pump

2



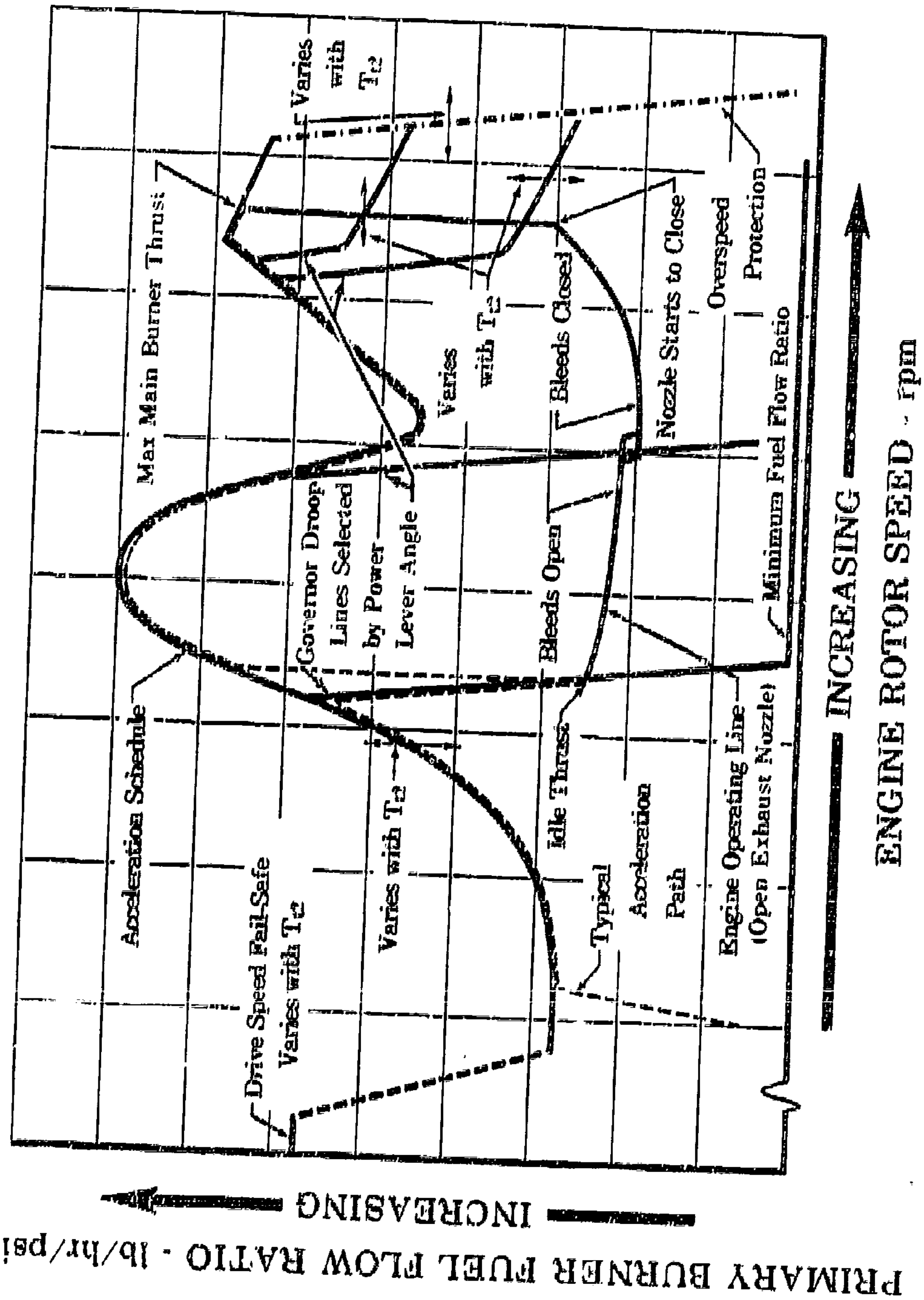

STJ227 MAIN GEARBOX

Figure 2B-82



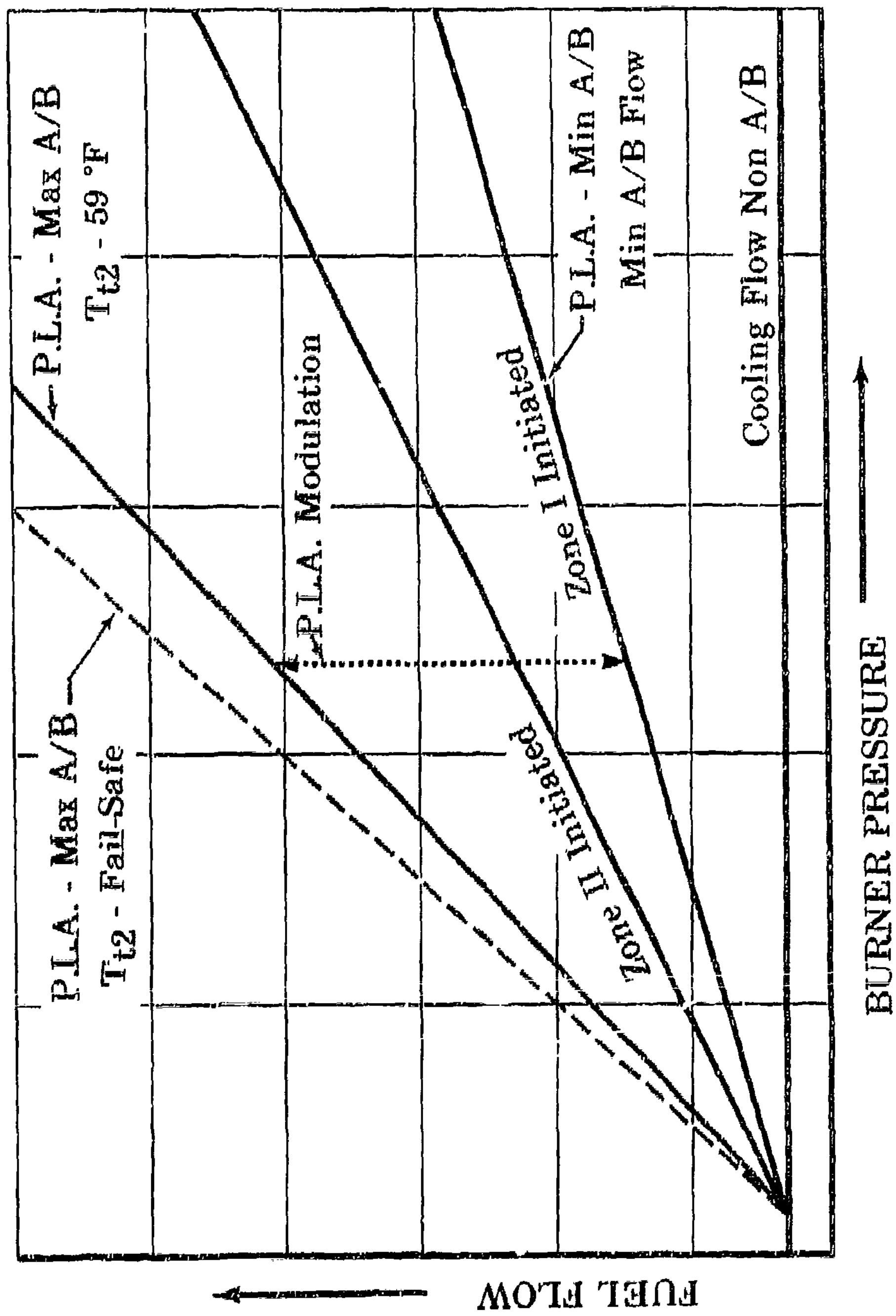

POWER LEVER OPERATION

Figure 2B-83



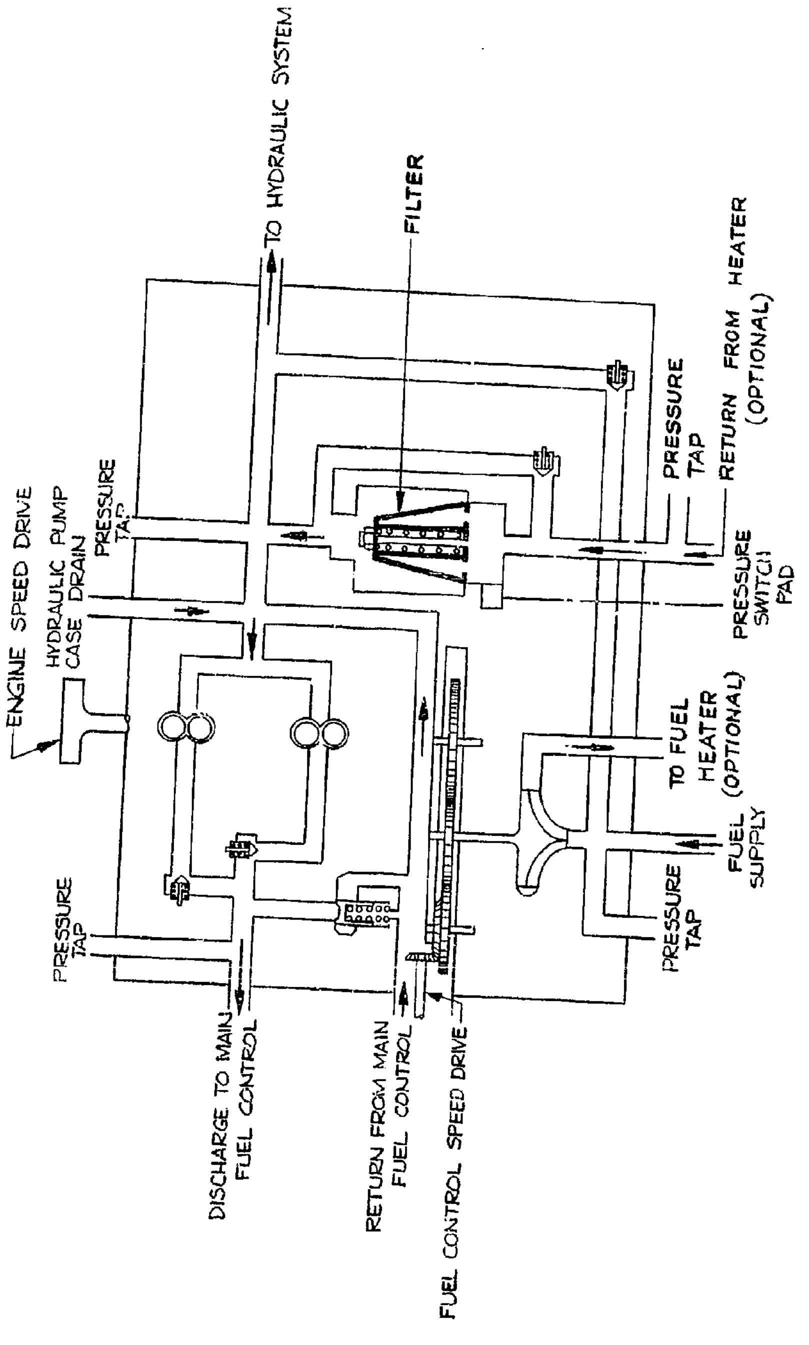

MAIN BURNER FUEL SCHEDULE

Figure 2B-84



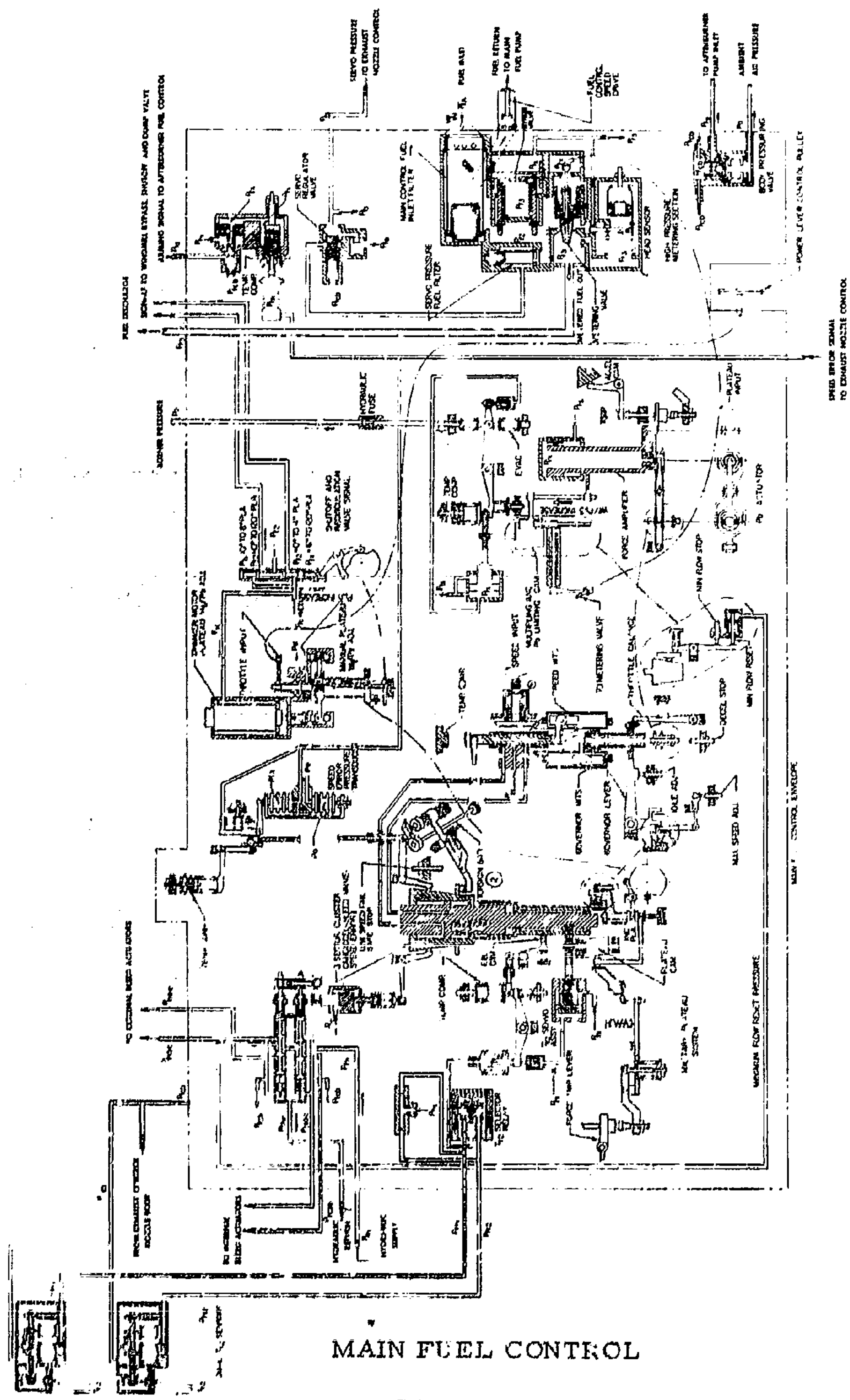

AFTERBURNER FUEL SCHEDULE

Figure 2B-85



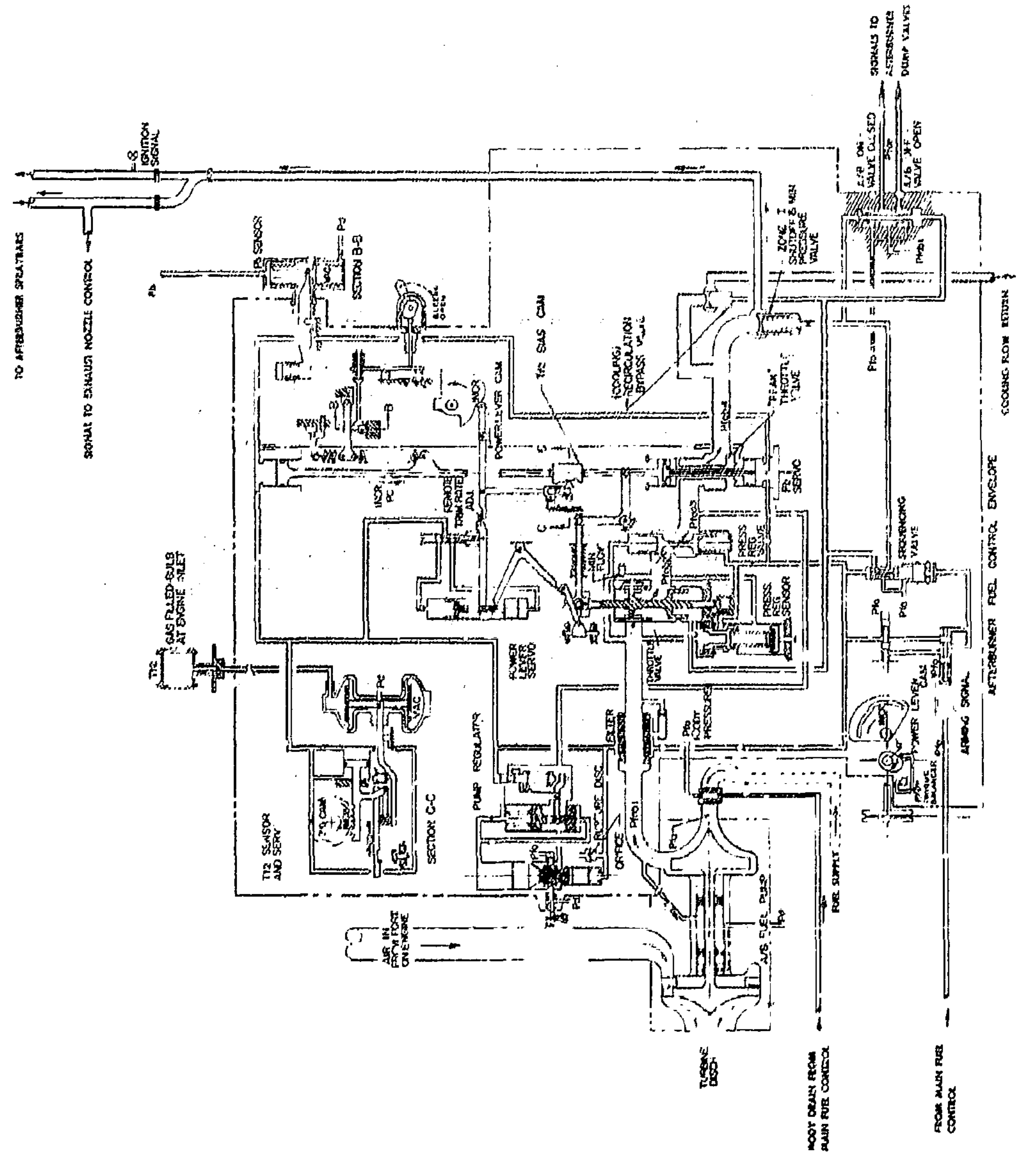

MAIN FUEL PUMP FLOW PASSAGES

Figure 2B-86



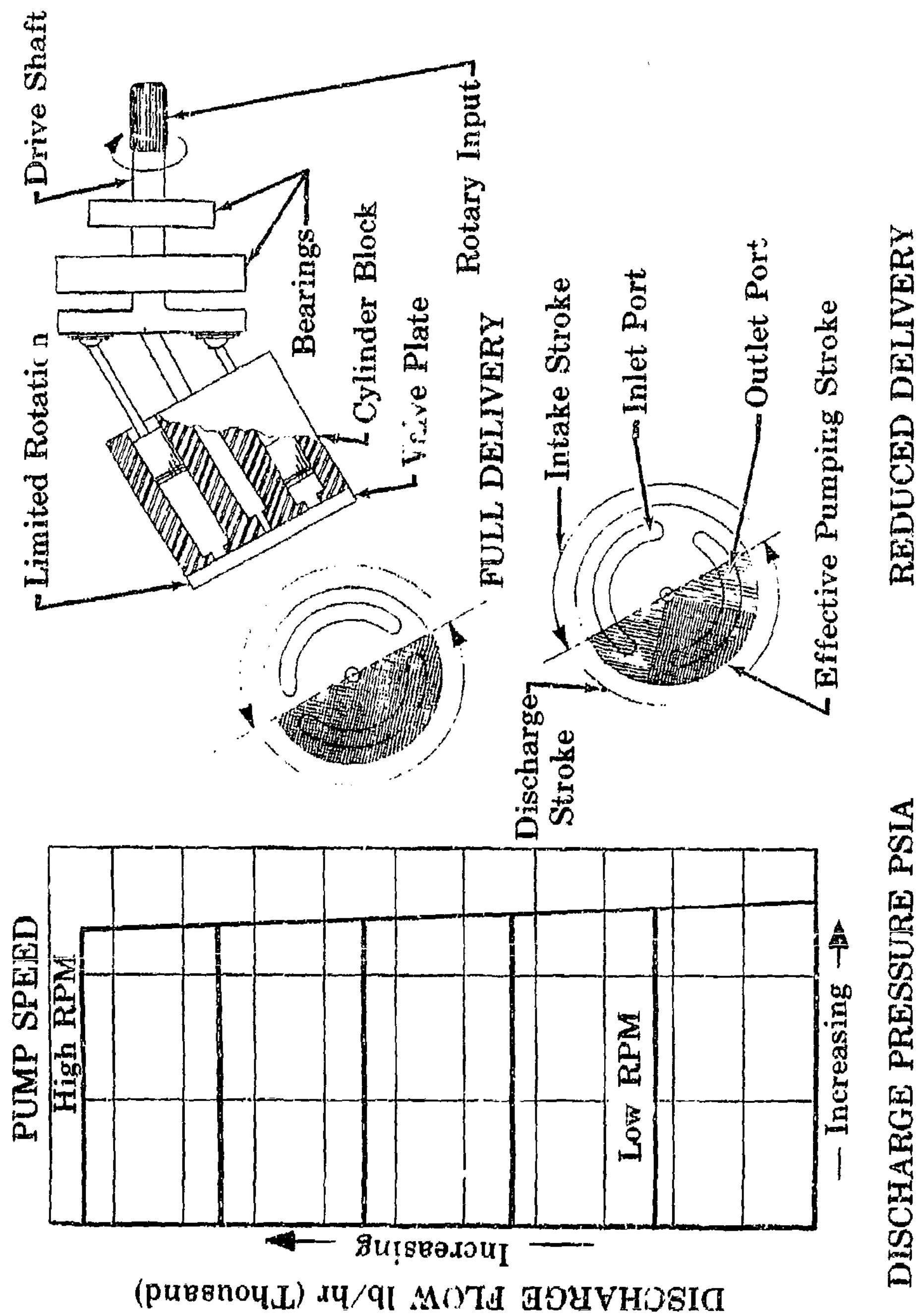
MAIN FUEL CONTROL

Figure 2B-87



AFTERL 'RNER FUEL CONTROL AND PUMP

Figure 2B-88




HYDRAULIC PUMP CHARACTERISTICS

Figure 2B-89

### ACCESSORY DRIVE PADS

| ZONE | | |
|---|---|---|
| 23F,17F, | A | POWER TAKEOFF |
| 30E | B | TACHOMETER MOUNTING PAD (N1)(AND 20005XV8) |

### FUEL DRAIN

| ZONE | | |
|---|---|---|
| 17C | FD1 | COMBUSTION CHAMBER FUEL DRAIN |
| 18C | FD2 | DUMP VALVE DRAIN (MAIN) |
| 15C | FD4 | DUMP VALVE DRAIN (AFTERBURNER)ZONE 1 |
| 16C | FD4A | DUMP VALVE DRAIN (AFTERBURNER)ZONE 2 |
| 13C | FD8 | AFTERBURNER COMBUSTION CHAMBER FUEL DRAIN |

### FUEL PRESSURE

| ZONE | | |
|---|---|---|
| 29E | FP1 | FUEL PUMP INLET PRESSURE |
| 30E | FP3 | CHECK VALVE FUEL PRESSURE (OUTLET PRESSURE) |

### FUEL FLOW

| ZONE | | |
|---|---|---|
| 19F, 30F | F1 | FUEL PUMP & AFTERBURNER PUMP SUPPLY INLET |
| 30D | F2 | MAIN FUEL FLOWMETER SUPPLY INLET |
| 30D | F3 | MAIN FUEL FLOWMETER SUPPLY OUTLET |
| 19E | F5 | AFTERBURNER FLOWMETER SUPPLY INLET |
| 19D | F6 | AFTERBURNER FLOWMETER SUPPLY OUTLET |

### FUEL VENT

| ZONE | | |
|---|---|---|
| 29D | FV2 | FUEL PUMP OUTLET VENT |

### FUEL TEMPERATURE

| ZONE | | |
|---|---|---|
| 30E | FT1 | HEATER OUTLET FUEL TEMPERATURE |

### OIL BREATHER

| ZONE | | |
|---|---|---|
| 30D | LB2 | MAIN OIL OVERBOARD BREATHER |

### OIL DRAIN

| ZONE | | |
|---|---|---|
| 29C | LD1 | OIL TANK DRAIN |
| 20D | LD2 | OIL TANK OVERFLOW DRAIN |
| 29D | LD3 | OIL CUP OVERFLOW DRAIN |
| 25C | LD4 | GEARBOX MAIN OIL DRAIN |
| 30C | LD7 | OIL STRAINER DRAIN |

### OIL FLOW

| ZONE | | |
|---|---|---|
| 29E | L1 | OIL TANK REMOTE FILLER |
| 29E | L2 | OIL TANK MANUAL FILL |

### OIL PRESSURE

| ZONE | | |
|---|---|---|
| 31D | LP1 | PRESSURE FOR TRANSMITTER |
| 25D | LP4 | OIL FILTER INLET PRESSURE } PROVISIONS FOR PRESSURE DIFFERENTIAL |
| 25C | LP5 | OIL FILTER OUTLET PRESSURE } PROVISIONS FOR PRESSURE DIFFERENTIAL |

### OIL TEMPERATURE

| ZONE | | |
|---|---|---|
| 31D | LT1 | MAIN OIL TEMPERATURE |

### OIL VENT

| ZONE | | |
|---|---|---|
| 30F | LV3 | OIL PRESSURE TRANSMITTER VENT |

### SEAL DRAIN

| ZONE | | |
|---|---|---|
| 29D | SD1 | FUEL CONTROL SEAL DRAIN |
| 30E | SD2 | FUEL PUMP SEAL DRAIN |
| 30D | SD9 | HYDRAULIC PUMP SEAL DRAIN |
| 19D | SD14 | AFTERBURNER FUEL PUMP SEAL DRAIN |

### TEMPERATURE SENSING

| ZONE | | |
|---|---|---|
| 20D | TT5 | TURBINE EXIT TEMPERATURE (AVG) |
| 20D | ITT5 | TURBINE EXIT TEMPERATURE (INDIVIDUAL) |

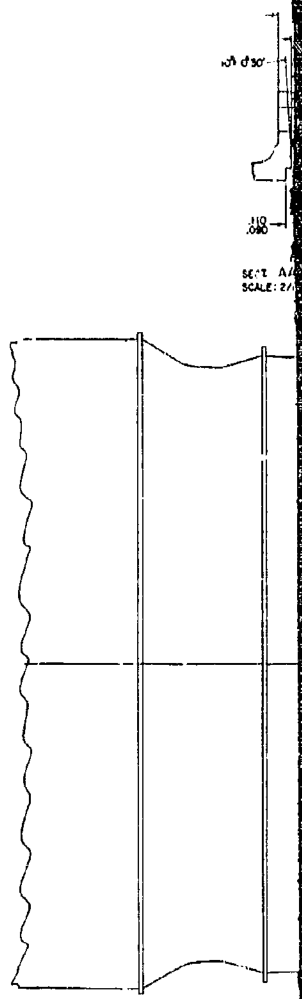
### PRESSURE SENSING

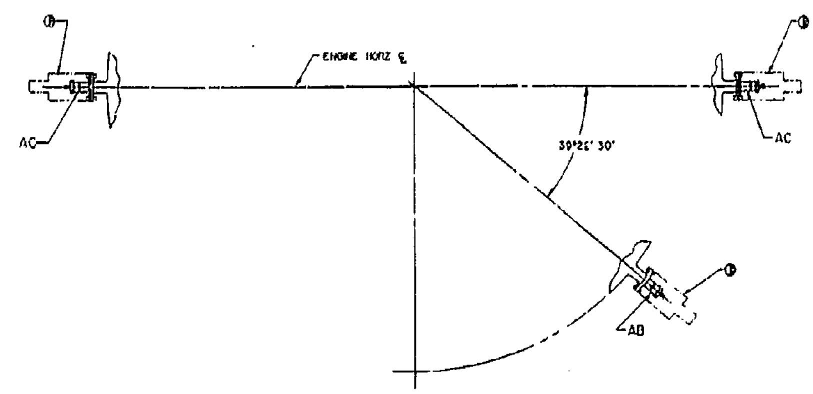
| ZONE | | |
|---|---|---|
| 15F | PT5 | TURBINE EXIT PRESSURE |

### MISCELLANEOUS

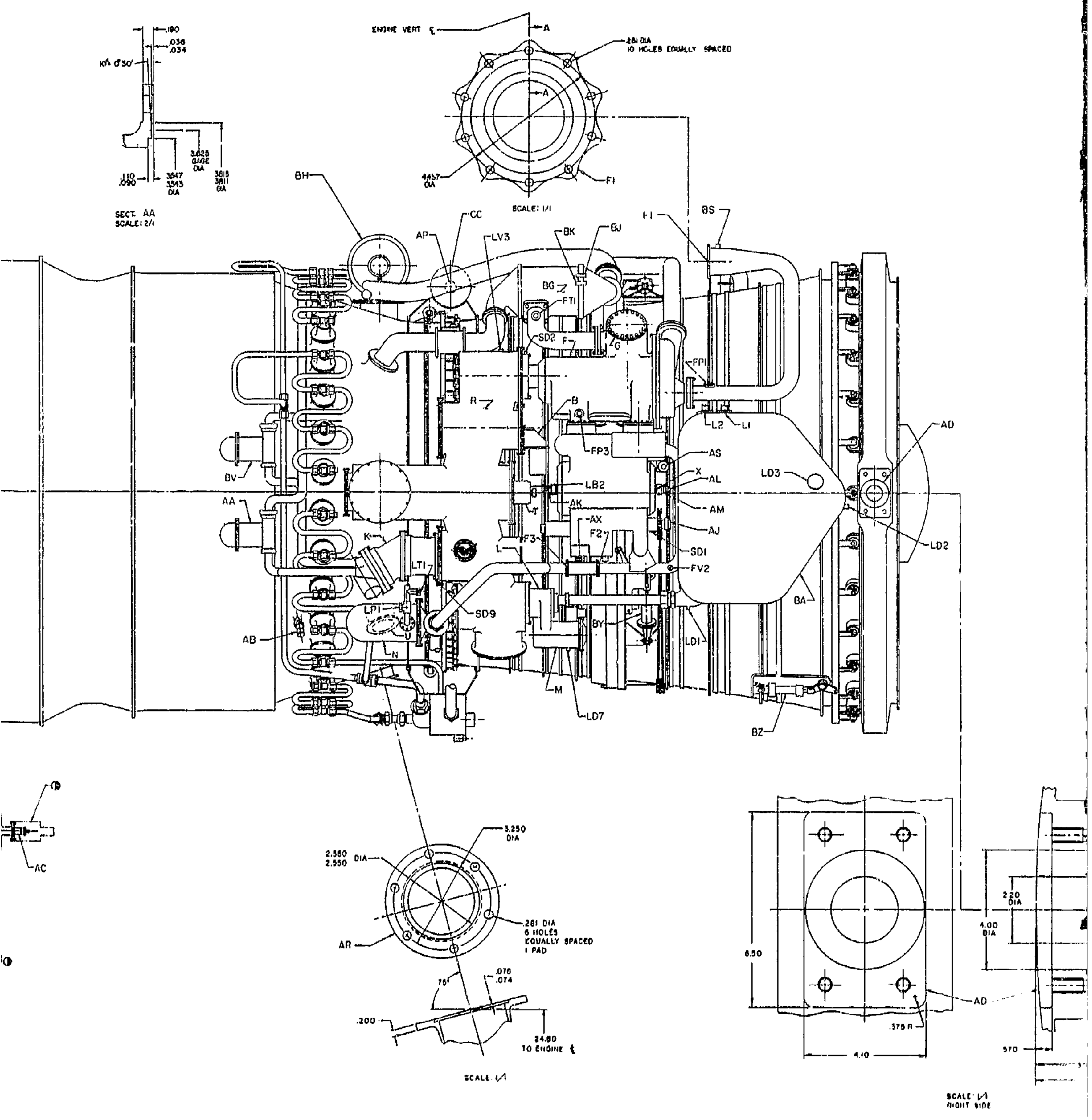
| ZONE | | |
|---|---|---|
| 26E, 30E | F | MAIN FUEL PUMP |
| 25E, 30E | G | FUEL PUMP FILTER (MAIN) |
| 20E | H | FUEL PUMP FILTER (MIN SPACE FOR REMOVAL) |
| 19E, 23E | J | AFTERBURNER FUEL PUMP |
| 31D | K | [illegible] PUMP |
| 28C, 30D | L | [illegible] |
| 26C, 30C | M | [illegible] |
| 31C | N | FUEL OIL COOLER (MAIN) |
| 19C, 23C | P | FUEL OIL COOLER (AFTERBURNER) |
| 30E | R | [illegible] |
| 20D | S | [illegible] RESTART SWITCH |
| 30D | T | [illegible] |
| 18E, 19D, 31C | U | [illegible] |
| 26D, 29D | X | MAIN FUEL CONTROL |
| 18D, 23D | Y | AFTERBURNER FUEL CONTROL |
| 17D | Z | EXHAUST NOZZLE CONTROL |
| 32D | AA | HYDRAULIC DISCHARGE FILTER |
| 32D, 34D, [illegible] | AB | IGNITER PLUG (MAIN) |
| 15D, 33D, 36D | AC | IGNITER PLUG (AFTERBURNER) |
| 21D, 21E, [illegible] | AD | ENGINE FRONT MOUNTING PROVISIONS |
| 13D, 16D, 16E | AE | ENGINE REAR MOUNTING PROVISIONS |
| 20F | AF | FUEL RETURN TO SUPPLEMENTARY COOLING |

| ZONE | | |
|---|---|---|
| 23D | AG | GROUND HANDLING HOLES (FRONT MOUNT) |
| 14C | AH | GROUND HANDLING AREA (REAR) |
| 26D | AJ | POWER CONTROL LEVER (120° ANGLE OF TRAVEL) |
| 30D | AK | SHUT OFF LEVER (90° ANGLE OF TRAVEL) |
| 26D | AL | APPROACH VELOCITY CONTROL LEVER (60° ANGLE OF TRAVEL) |
| 26D, 29D | AM | MACH NO. OR SHOCK POSITION RESET LEVER (60° ANGLE OF TRAVEL) |
| 18E | AN | AIR TURBINE VALVE AFTERBURNER FUEL PUMP |
| 31F | AP | IGNITION EXCITER ELECTRICAL CONN. |
| 31D | AR | THERMAL ANTI ICING PAD |
| 25E, 29D | AS | AERODYNAMIC BRAKE CONTROL AIR SUPPLY CONN. |
| 17D | AT | EXHAUST NOZZLE FEEDBACK |
| 14C | AU | CHECK & DUMP VALVE (AFTERBURNER ZONE I) |
| 16C | AV | CHECK & DUMP VALVE (AFTERBURNER ZONE II) |
| 19D | AW | MAIN FUEL FLOWMETER MOUNTING PROVISIONS |
| 30D | AX | AFTERBURNER FUEL FLOWMETER MOUNTING PROVISIONS |
| 23E, 19E | AY | FUEL CONTROL FUEL FILTER (MIN SPACE FOR REMOVAL) |
| 18C, 25C | AZ | WINDMILL BYPASS CHECK & DUMP LOW IDLE VALVE (MAIN) |
| 29D | BA | OIL TANK |
| 17D | BB | NOZZLE POSITION INDICATOR MOUNTING PROVISIONS |
| 20D | BC | REVERSER POSITION INDICATOR CONN. (ELECTRICAL) |
| 19C | BD | THERMOSTATIC FUEL BYPASS VALVE |
| 16A, 22B, 16D, 21D | BF | VIBRATION MOUNT (SPACE RESERVED FOR PICKUP PROVIDED BY AIRFRAME MFG) |
| 30E | BG | FUEL HEATER |
| 17F, 24F, 31F | BH | POWER TAKEOFF GEARBOX |
| 30F | BJ | FUEL HEATER VALVE (ELECTRICAL CONN) |
| 30F | BK | FUEL HEATER VALVE POSITION INDICATOR (ELECTRICAL) |
| 17F | BL | POWER TAKEOFF DECOUPLER |
| 10F, 24F | BM | POWER TAKEOFF DECOUPLER ACTIVATING SHAFT |
| 24D | BN | CONTROL CABLE – FUEL CONTROLS |
| 19F, 29F | BS | FUEL INLET TEMPERATURE SENSOR |
| 18D | BU | AFTERBURNER FUEL OIL COOLER THERMOSTATIC BYPASS VALVE |
| 32D | BV | HYDRAULIC RETURN FILTERS |
| 14E | BW | EXHAUST NOZZLE ACTUATORS |
| 23D, 19D, 19F, 25F, 30C | BY | STARTING BLEED DOOR ACTUATORS |
| 20E, 29C | BZ | INLET GUIDE VANE ACTUATORS |
| 18C, 18F | CA | AERODYNAMIC BRAKE ACTUATORS |
| 20D | CB | EXCITER (MAIN) |
| 18F, 30F | CC | EXCITER AFTERBURNER |



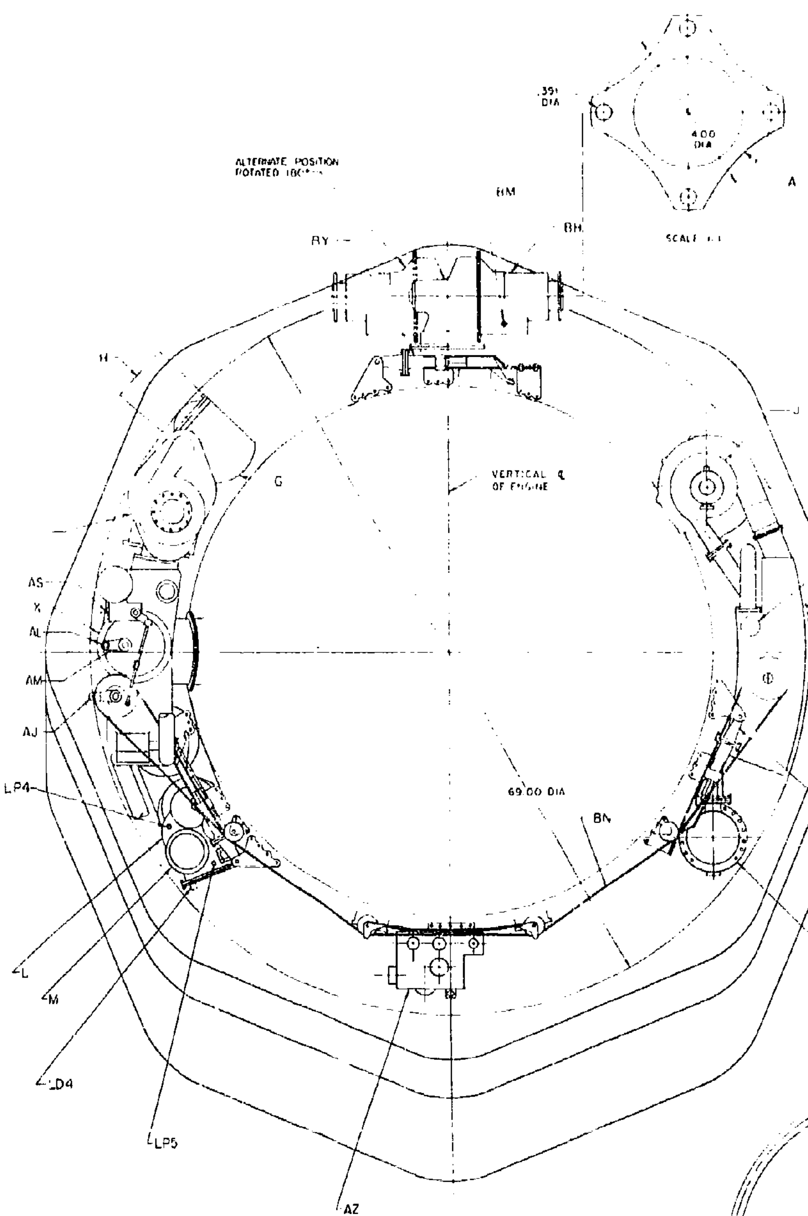




VIEW OF MAIN & AFTERBURNER IGNITER
LOOKING FORWARD

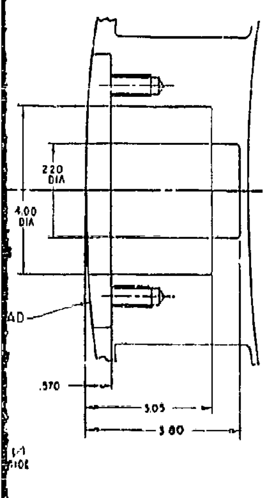


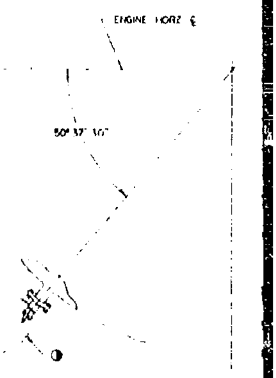
ENGINE VERT ℄
.281 DIA
10 HOLES EQUALLY SPACED
4.457 DIA
SCALE: 1/1
SECT. AA
SCALE: 2/1
10° 0'30'
.190
.036
.034
3.625 GAGE DIA
3.547
3.543 DIA
3.815
3.811 DIA
.110
.090
BH
CC
AP
LV3
BK
BJ
BG
FT1
F1
BS
SD2
F
G
FP1
R
B
L2
L1
AD
FP3
AS
X
AL
LD3
LB2
AM
AK
T
AX
AJ
F3
F2
SD1
LD2
L
K
LT1
FV2
BV
AA
LP1
SD9
BA
AB
U
N
BY
LD1
M
LD7
BZ
AC
3.250 DIA
2.560
2.550 DIA
.281 DIA
6 HOLES
EQUALLY SPACED
1 PAD
AR
.078
.074
75°
.200
24.80
TO ENGINE ℄
SCALE 1/1
6.50
4.10
.375 R
2.20 DIA
4.00 DIA
.570
SCALE 1/1
RIGHT SIDE


2

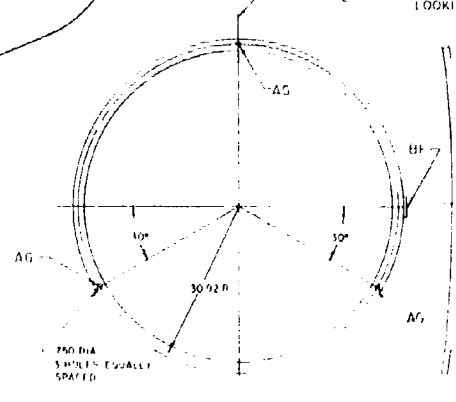



FRONT VIEW
SHOWING ACCESSORIES

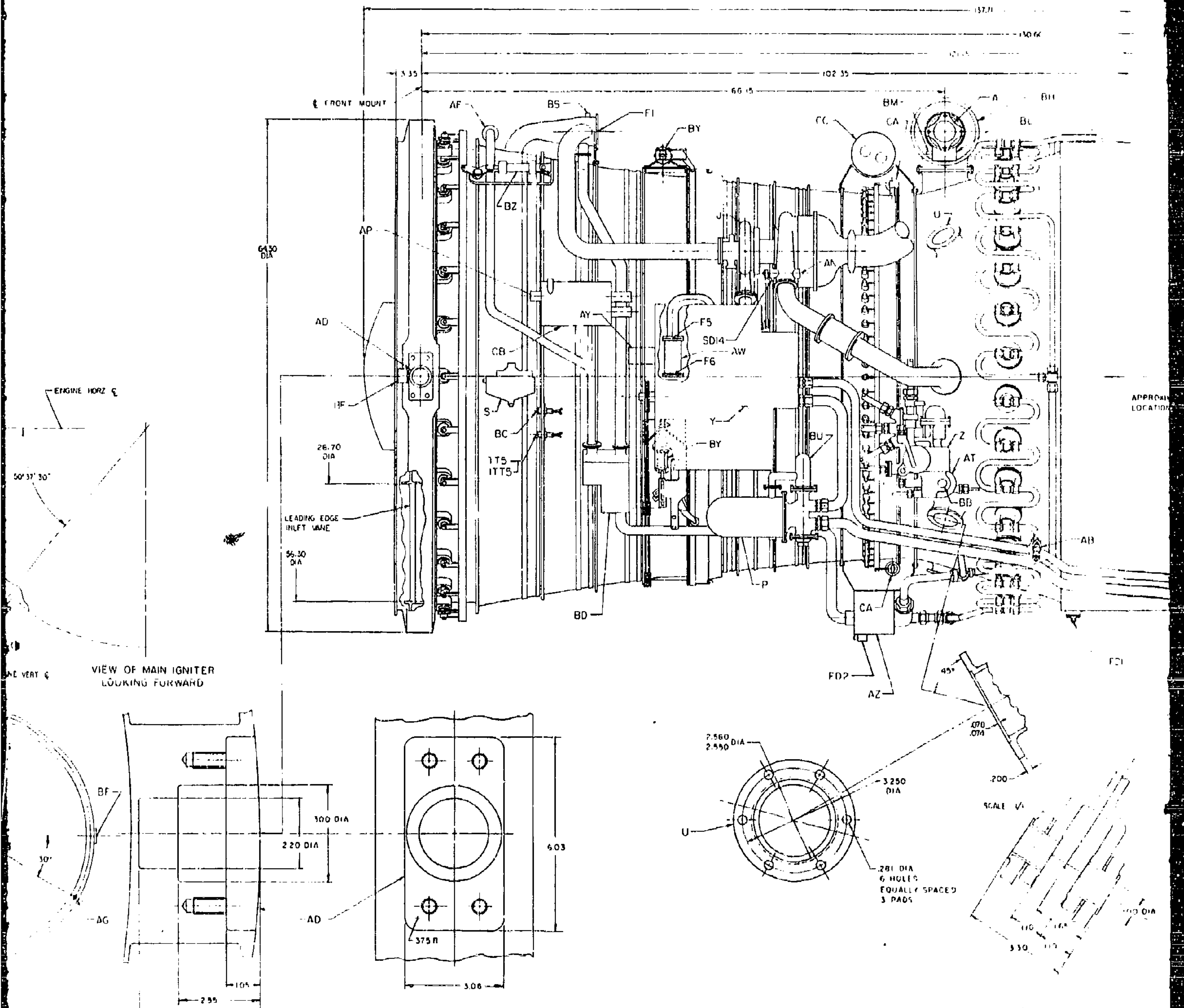



VIEW OF MAIN
LOOKING FO

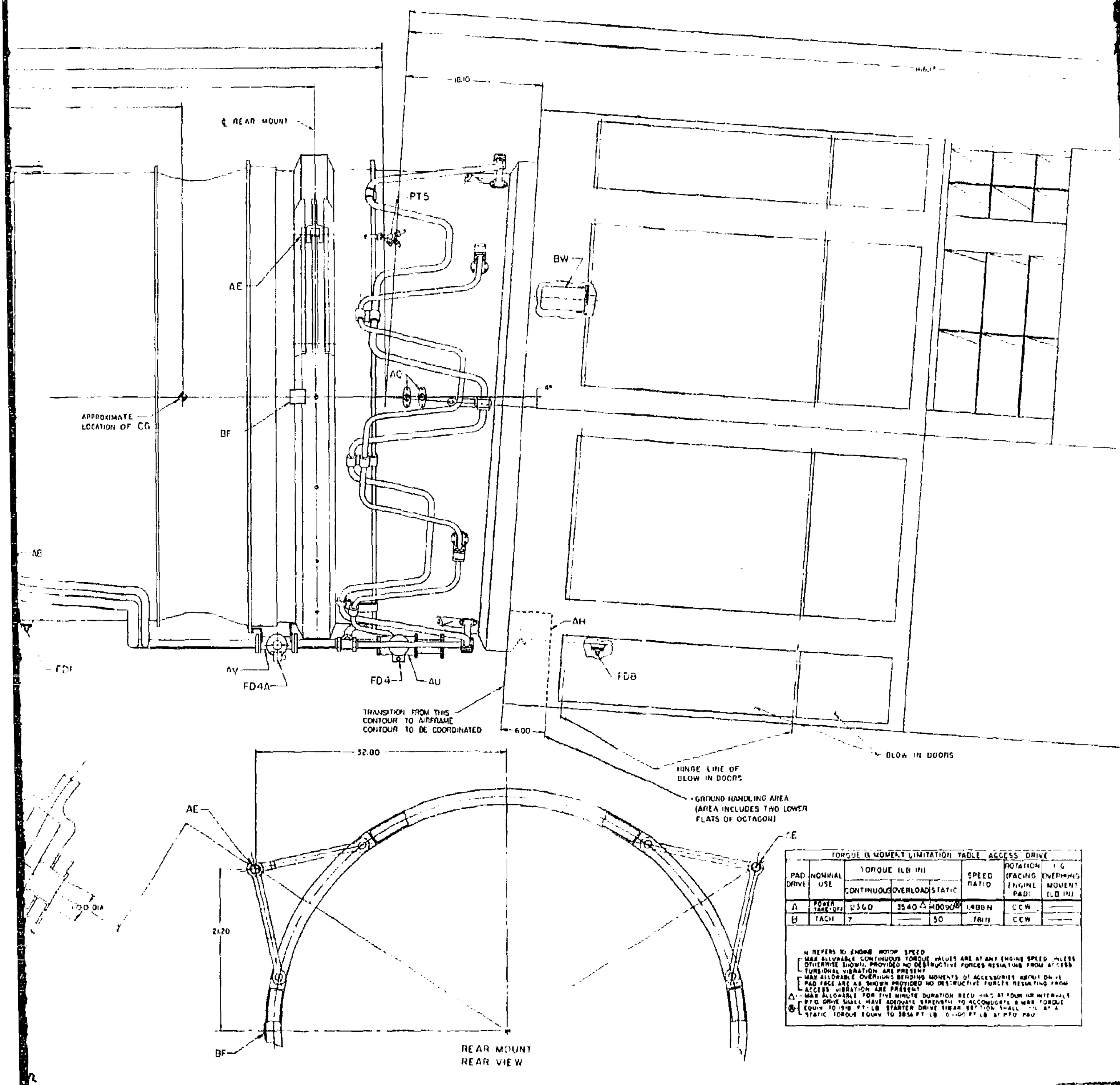



SCALE 1/6
FRONT MOUNT
FRONT VIEW



137.71
130.66
102.35
66.15
3.35
℄ FRONT MOUNT
64.30 DIA
AF
BS
F1
BY
CC
BM
CA
A
BH
BL
BZ
AP
AN
AY
F5
SD14
AW
F6
AD
CB
BF
S
BC
1T5
1TT5
Y
BY
BU
Z
AT
BB
AB
P
BD
CA
FD2
AZ
ENGINE HORZ ℄
26.70 DIA
LEADING EDGE INLET VANE
56.30 DIA
APPROX LOCATION
VIEW OF MAIN IGNITER LOOKING FORWARD
BF
AG
AD
3.00 DIA
2.20 DIA
6.03
.375 R
3.06
1.05
2.55
2.560 DIA
2.550
3.250 DIA
U
.281 DIA 6 HOLES EQUALLY SPACED 3 PADS
45°
.070
.074
.200


4

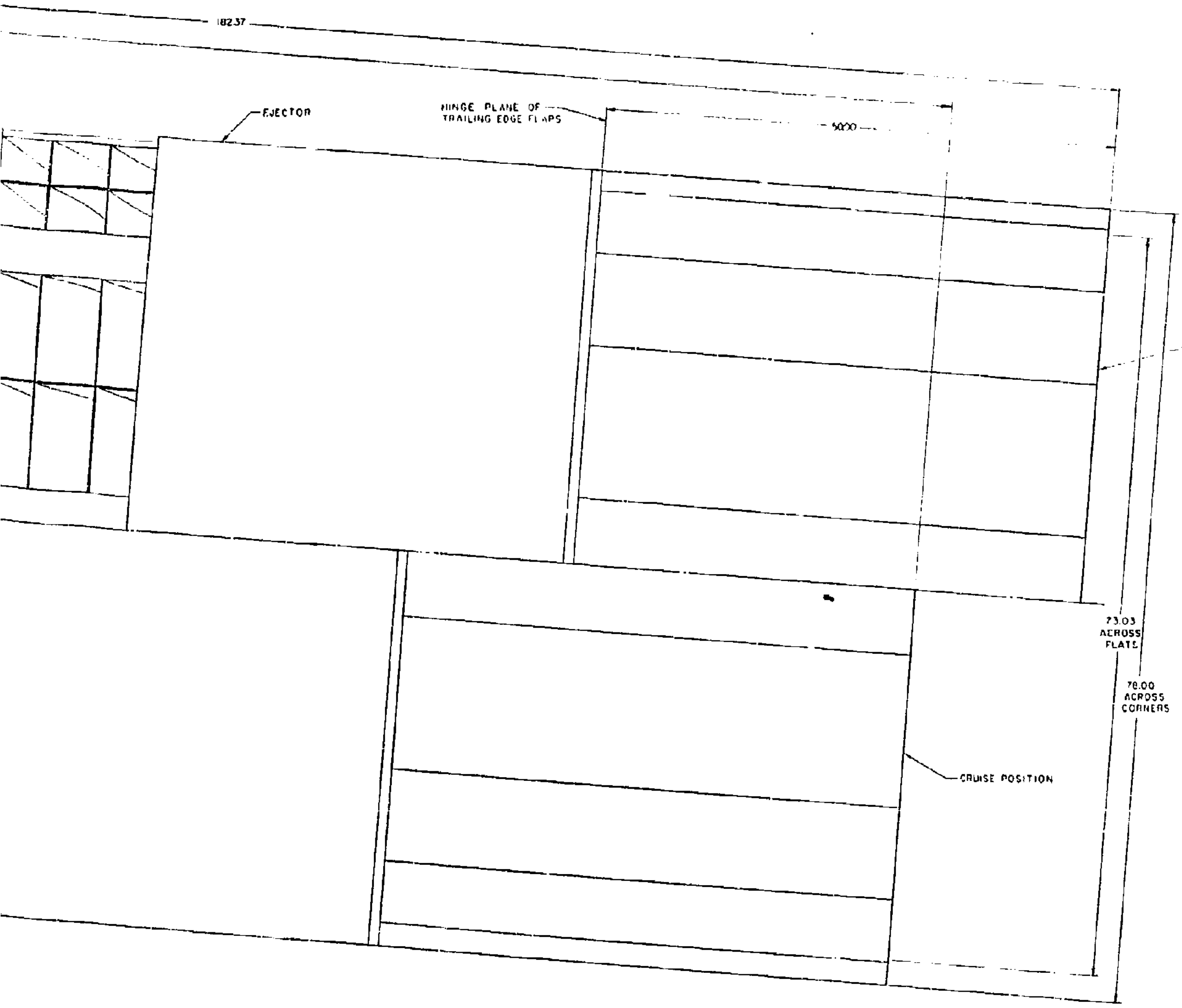



TORQUE & MOMENT LIMITATION TABLE ACCESS DRIVE

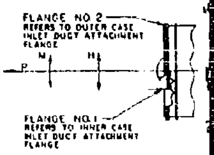
| PAD DRIVE | NOMINAL USE | TORQUE (LB IN) CONTINUOUS | TORQUE (LB IN) OVERLOAD | TORQUE (LB IN) STATIC | SPEED RATIO | ROTATION (FACING ENGINE PAD) | I G OVERHUNG MOMENT (LB IN) |
|---|---|---|---|---|---|---|---|
| A | POWER TAKE-OFF | 2360 | 3540 △ | 40090 ® | 1.400 N | CCW | —— |
| B | TACH | 7 | —— | 50 | .781 N | CCW | —— |

N REFERS TO ENGINE ROTOR SPEED
MAX ALLOWABLE CONTINUOUS TORQUE VALUES ARE AT ANY ENGINE SPEED UNLESS OTHERWISE SHOWN, PROVIDED NO DESTRUCTIVE FORCES RESULTING FROM ACCESS TORSIONAL VIBRATION ARE PRESENT
MAX ALLOWABLE OVERHUNG BENDING MOMENTS OF ACCESSORIES ABOUT DRIVE PAD FACE ARE AS SHOWN PROVIDED NO DESTRUCTIVE FORCES RESULTING FROM ACCESS VIBRATION ARE PRESENT
△ — MAX ALLOWABLE FOR FIVE MINUTE DURATION RECURRING AT FOUR HR INTERVALS
® — PTO DRIVE SHALL HAVE ADEQUATE STRENGTH TO ACCOMMODATE A MAX TORQUE EQUIV TO 1910 FT-LB STARTER DRIVE SHEAR SECTION SHALL FAIL AT A STATIC TORQUE EQUIV TO 3034 FT-LB (3400 FT LB AT PTO PAD)

5




LIMIT LOADS




(SUBSCRIPTS 1 &

MAX ALLOWABLE MA

1 [3.43 P1 + 19

2 [1.60 P1 + 32

MAX ALLOWABLE M

3 [1.19 (P1 + P2)

MAX ALLOWABLE 1.0

4 M1 + M2 & 720

MAX LOAD PER HOLE

(ALL LOADS AND MO

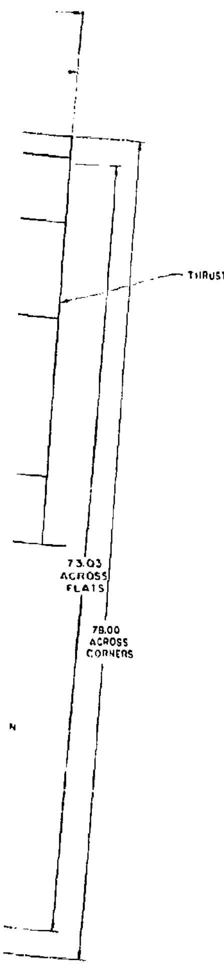
| TABLE ACCESS DRIVE | | | |
|---|---|---|---|
| | SPEED RATIO | ROTATION (FACING ENGINE PAD) | 1 G OVERHUNG MOMENT (LB IN) |
| | 1.408N | CCW | —— |
| | .781N | CCW | —— |

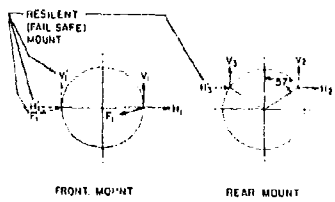
ARE AT ANY ENGINE SPEED UNLESS
FORCES RESULTING FROM ACCESS

OF ACCESSORIES ABOUT DRIVE
UCTIVE FORCES RESULTING FROM

RECURRING AT FOUR HR INTERVALS
TO ACCOMODATE A MAX TORQUE
IN SECTION SHALL FAIL AT A
IN-LB AT PTO PAD

6

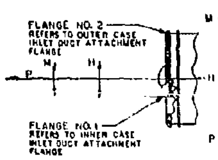



| | |
|---|---|
| \|V1\| ≤ 10000 LBS | \|V1'\| ≤ 10000 LBS |
| \|H1\| ≤ 25000 LBS | \|H1'\| ≤ 25000 LBS |
| \|F1\| ≤ 80000 LBS | \|F1'\| ≤ 80000 LBS |
| \|V2\| ≤ 35000 LBS | \|H3'\| ≤ 40000 LBS |
| \|H2\| ≤ 40000 LBS | |
| \|V3\| ≤ 40000 LBS | |

| | THESE LOADS REPRESENT THE MAX LOADS AT EACH LOCATION AND ARE NOT COMPATIBLE THEY DO NOT OCCUR SIMULTANEOUSLY

LIMIT LOADS FOR ENGINE INLET FLANGE




M = TOTAL MOMENT IN PLANE CONTAINING ENGINE AXIS THAT IS APPLIED TO FLANGE NO. 1 OR NO. 2 DUE TO INSTALLATION LOADS AND MANEUVER LOADS (IN-LB)

H = TOTAL LOAD PERPENDICULAR TO ENGINE AXIS (SHEAR) THAT IS APPLIED TO FLANGE NO 1 OR NO 2 DUE TO INSTALLATION LOADS AND MANEUVER LOADS (LBS)

P = TOTAL AXIAL LOAD APPLIED TO FLANGE NO. 1 OR NO 2 DUE TO INSTALLATION LOADS AND MANEUVER LOADS (LBS)

(SUBSCRIPTS 1 AND 2 REFER TO FLANGES 1 AND 2 RESPECTIVELY)

MAX ALLOWABLE MANEUVER LOAD EQUATIONS FOR FLANGE NO. 1

1. $[3.43 P_1 + .79 M_1]\ 10^{-3} \leq 1$
2. $[1.60 P_1 + .32 H_1 + .49 M_1]\ 10^{-3} \leq 1$

MAX ALLOWABLE MANEUVER LOAD EQUATION FOR FLANGES NO. 1 AND 2

3. $[1.19\ (P_1 + P_2) + .11\ (M_1 + M_2)]\ 10^{-3} \leq 1$

MAX ALLOWABLE 1.0 (GROUND) LOAD EQUATIONS

4. $M_1 + M_2 \leq 720$ IN-LBS. $H_1 + H_2 \leq 600$ LBS. $P_1 + P_2 \leq 600$ LBS.

MAX LOAD PER HOLE AT FRONT FACE OF FLANGE ≤ 140 LBS
(ALL LOADS AND MOMENTS TO BE USED AS ABSOLUTE VALUES)

PRELIMINARY INSTALLATION DWG
PRINTED JUNE 17, 1965

(4)— (A) MIN SPACE FOR REMOVAL
(3)— DATUMS FOR TRUE POSITION OF FEATURES SHOWN ARE CENTERLINES OF CASE CONTAINING SUCH FEATURES UNLESS OTHERWISE INDICATED
(2)— ALL DIMENSIONS SHOWN ARE NOMINAL UNLESS OTHERWISE ...
(1)— ALL DIMENSIONS GIVEN FOR ROOM TEMPERATURE

STJ227L ENGINE INSTALLATION (LOCKHEED)

Figure 2B-90



7

## ACCESSORY DRIVE PADS

| ZONE | | |
|---|---|---|
| 42, 71, 30E | A | POWER TAKEOFF |
| 300 | B | TACHOMETER MTG PAD ONLY (AND 20005XVIII) |
| 30E 300 | D | HYDRAULIC PUMP (TYPE) (AS 418(I)) |

## FUEL DRAIN

| | | |
|---|---|---|
| 17B | FD1 | COMBUSTION CHAMBER FUEL DRAIN |
| 19B, 30C | FD2 | DUMP VALVE DRAIN (MAIN) |
| 15B | FD4 | DUMP VALVE DRAIN (A/B ZONE I) |
| 16B | FD4A | DUMP VALVE DRAIN (A/B ZONE II) |
| 13B | FD8 | AFTERBURNER COMBUSTION CHAMBER FUEL DRAIN |

## FUEL PRESSURE

| | | |
|---|---|---|
| 29F | FP1 | FUEL PUMP INLET PRESSURE |
| 30E | FP3 | CHECK VALVE FUEL PRESSURE (OUTLET PRESSURE) |

## FUEL FLOW

| | | |
|---|---|---|
| 19E | F1 | FUEL PUMP & AFTERBURNER PUMP SUPPLY INLET |
| 300 | F2 | MAIN FUEL FLOWMETER SUPPLY INLET |
| 300 | F3 | MAIN FUEL FLOWMETER SUPPLY OUTLET |
| 19E | F5 | AFTERBURNER FLOWMETER SUPPLY INLET |
| 190 | F6 | AFTERBURNER FLOWMETER SUPPLY OUTLET |

## FUEL VENT

| | | |
|---|---|---|
| 290 | FV2 | FUEL PUMP OUTLET VENT |

## PRESSURE SENSING

| | | |
|---|---|---|
| [illegible] | PT5 | TURBINE EXIT PRESSURE |

## OIL BREATHER

| | | |
|---|---|---|
| 300 | LB2 | MAIN OIL OVERBOARD BREATHER |

## OIL DRAIN

| | | |
|---|---|---|
| 29C | LD1 | OIL TANK DRAIN |
| 28D | LD2 | OIL TANK OVERFLOW DRAIN |
| 28D | LD3 | OIL CUP OVERFLOW DRAIN |
| 25C | LD4 | GEARBOX MAIN OIL DRAIN |
| 29B | LD7 | OIL STRAINER DRAIN |

## OIL FLOW

| | | |
|---|---|---|
| 290 | L1 | OIL TANK REMOTE FILLER |
| 290 | L2 | OIL TANK MANUAL FILL |

## OIL PRESSURE

| | | |
|---|---|---|
| 3B | LP1 | PRESSURE FOR TRANSMITTER |
| 25C | LP4 | OIL FILTER INLET PRESSURE |
| 25C | LP5 | OIL FILTER OUTLET PRESSURE |

LP4, LP5 — PROVISIONS FOR PRESSURE DIFFERENTIAL

## OIL TEMPERATURE

| | | |
|---|---|---|
| 300 | LT1 | MAIN OIL TEMPERATURE |

## OIL VENT

| | | |
|---|---|---|
| 30E | LV3 | OIL PRESSURE TRANSMITTER VENT |

## SEAL DRAIN

| | | |
|---|---|---|
| 290 | SD1 | FUEL CONTROL SEAL DRAIN |
| 10C | SD2 | FUEL PUMP SEAL DRAIN |
| 30E | SD9 | HYDRAULIC PUMP SEAL DRAIN |
| 19D | SD14 | AFTERBURNER FUEL PUMP SEAL DRAIN |

## TEMPERATURE SENSING

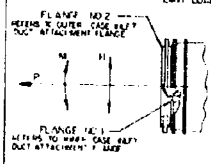
| | | |
|---|---|---|
| 20E | TT5 | TURBINE EXIT TEMPERATURE (AVG) |
| 20F | ITT5 | TURBINE EXIT TEMPERATURE (INDIVIDUAL) |

## MISCELLANEOUS

| | | |
|---|---|---|
| [illegible] | F | MAIN FUEL PUMP |
| 29E 29F | G | FUEL PUMP FILTER (MAIN) |
| 29E | H | FUEL PUMP FILTER (SPACE FOR REMOVAL) |
| 18E | J | AFTERBURNER FUEL PUMP |
| 31D | K | HYDRAULIC PUMP |
| 25C 30C | L | OIL PUMP |
| 25C 30C | M | OIL FILTER |
| 30C | N | FUEL OIL COOLER (MAIN) |
| 23C | P | FUEL OIL COOLER (AFTERBURNER) |
| 30E | R | GEARBOX |
| 25C | S | AUTOMATIC RESTART SWITCH |
| 30E | T | BREATHER PRESSURIZING VALVE |
| [illegible] | U | HIGH PRESSURE BLEED PAD |
| 29D 29C | X | MAIN FUEL CONTROL |
| [illegible] | Y | AFTERBURNER FUEL CONTROL |

| ZONE | | |
|---|---|---|
| 17D | Z | EXHAUST NOZZLE CONTROL |
| 32D | AA | HYDRAULIC DISCHARGE FILTER |
| [illegible] | AB | IGNITER PLUG (MAIN) |
| [illegible] | AC | IGNITER PLUG (AFTERBURNER) |
| 21F 22C | AD | ENGINE FRONT MOUNTING PROVISIONS |
| 15E | AE | ENGINE REAR MOUNTING PROVISIONS |
| 20E 24E | AF | FUEL RETURN TO SUPPLEMENTARY COOLING |
| 20A 23A | AG | GROUND HANDLING HOLES (FRONT MOUNT) |
| 14C | AH | GROUND HANDLING AREA (REAR) |
| 250 290 | AJ | POWER CONTROL LEVER (120° ANGLE OF TRAVEL) |
| 30E | AK | SHUT OFF LEVER (90° ANGLE OF TRAVEL) |
| 29E | AL | AFTERBURNER VELOCITY CONTROL LEVER (ANGLE OF TRAVEL) |
| 29D | AM | MACH NO OR STOCK POSITION [illegible] LEVER [illegible] ANGLE OF TRAVEL |
| 19E | AN | AIR TURBINE VALVE (AFTERBURNER FUEL PUMP) |
| 2CL 30F | AP | IGNITION EXCITER ELECTRICAL CONN |
| 34 | AR | THERMAL ANTI ICING PAD |
| 29D | AS | AERODYNAMIC BRAKE CONTROL AIR SUPPLY CONN |
| 17C | AT | EXHAUST NOZZLE FEEDBACK |
| 15B | AU | CHECK & DUMP VALVE (AFTERBURNER ZONE I) |
| 15B | AV | CHECK & DUMP VALVE (AFTERBURNER ZONE II) |
| 19D | AW | GAS GENERATOR FUEL FLOWMETER |
| 29D | AX | MAIN FUEL FLOWMETER |
| 19D 22D | AY | AFTERBURNER FUEL FILTER |
| 18F 24B | AZ | WINDMILL BYPASS CHECK & DUMP VALVE (MAIN FUEL) |
| 28C | BA | OIL TANK |
| 17D | BB | NOZZLE POSITION INDICATOR MOUNTING PROVISIONS |
| 20D | BC | REVERSER POSITION INDICATOR CONN (ELECTRICAL) |
| 19C | BD | THERMOSTATIC FUEL BYPASS VALVE |
| 20A 6D 6A 23D | BF | VIBRATION MOUNT (SPACE RESERVED FOR PICKUP PROVIDED BY AIRFRAME MFG) |
| 23C | BN | CONTROL CABLE FUEL CONTROLS |
| 23C 23F 25C 28F | BP | SECONDARY AIR BYPASS [illegible] |
| 19E | BS | FUEL INLET TEMPERATURE SENSOR |
| 19D | BU | OIL COOLER THERMOSTATIC BYPASS VALVE AFTERBURNER FUEL |
| 31D | BV | HYDRAULIC RETURN FILTER |
| 14E | BW | EXHAUST NOZZLE ACTUATORS |
| 19D 19E 29C | BY | STARTING BLEED DOOR ACTUATORS |
| 28C | BZ | INLET GUIDE VANE ACTUATORS |
| 18C 18F | CA | AERODYNAMIC BRAKE ACTUATORS |
| 20D | CB | EXCITER MAIN |
| 18E 30E | CC | EXCITER (AFTERBURNER) |

### LIMIT LOADS FOR ENGINE INLET FLANGE




M = TOTAL MOMENT IN PLANE CONTAINING ENGINE AXIS THAT IS APPLIED TO FLANGE NO 1 OR NO 2 DUE TO INSTALLATION LOADS AND MANEUVER LOADS (IN LB)

H = TOTAL LOAD PERPENDICULAR TO ENGINE AXIS (@ EAR) THAT IS APPLIED TO FLANGE NO 1 OR NO 2 DUE TO INSTALLATION LOADS AND MANEUVER LOADS (LBS)

P = TOTAL AXIAL LOAD APPLIED TO FLANGE NO 1 OR NO 2 DUE TO INSTALLATION LOADS AND MANEUVER LOADS (LBS)

(SUBSCRIPTS 1 AND 2 REFER TO FLANGES 1 AND 2 RESPECTIVELY)

MAX ALLOWABLE MANEUVER LOAD EQUATIONS FOR FLANGE NO 1

1 $[3.43 P_1 + .79 M_1] 10^{-3} \le 1$

2 $[.60 P_1 + .32 H_1 + .49 M_1] 10^{-3} \le 1$

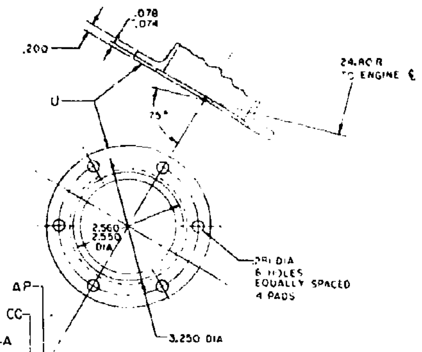
MAX ALLOWABLE 1 G (GROUND) LOAD EQUATIONS FOR FLANGE NO 2

3 { $M_2$ = [illegible] IN LBS $H_2$ = [illegible] LBS $P_2$ = [illegible] LBS

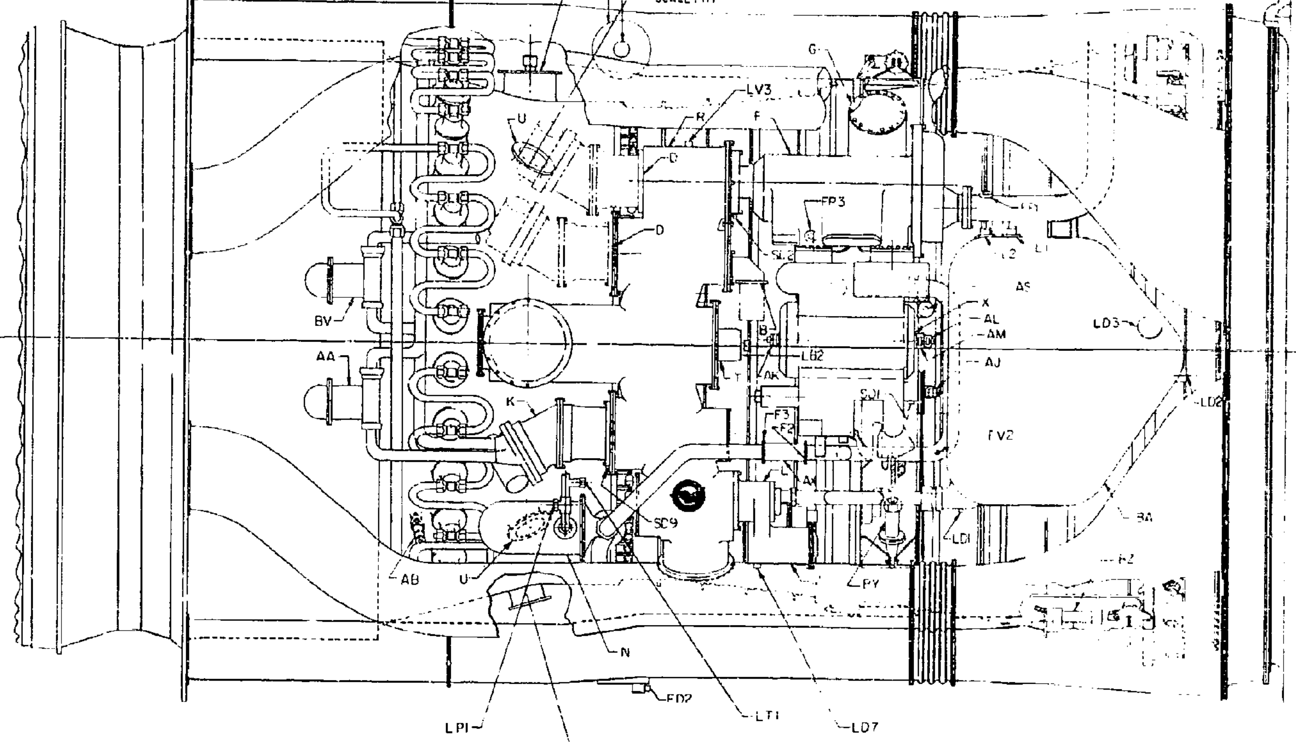
(ALL LOADS AND MOMENTS TO BE USED AS ABSOLUTE VALUES)

VIEW
LO

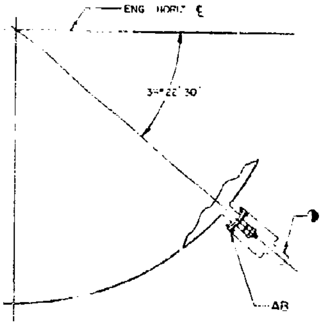
1



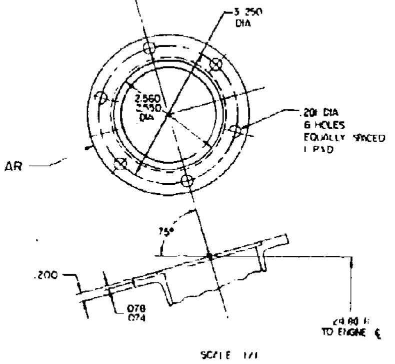

SCALE : 1/1




RIGHT SIDE VIEW

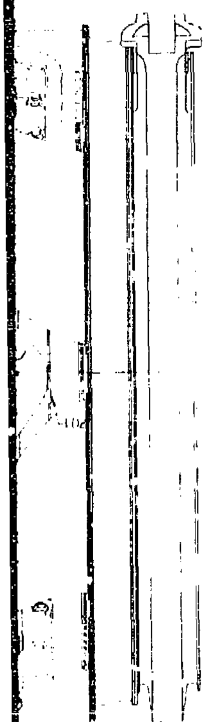


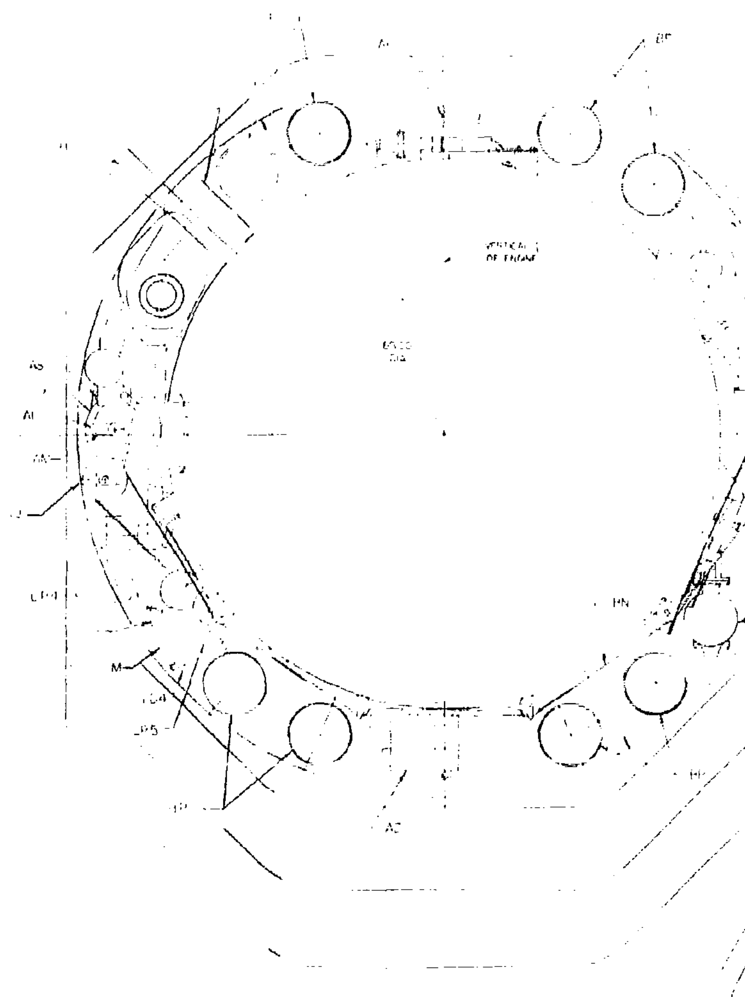

VIEW OF MAIN IGNITER
LOOKING FORWARD




SCALE 1/1

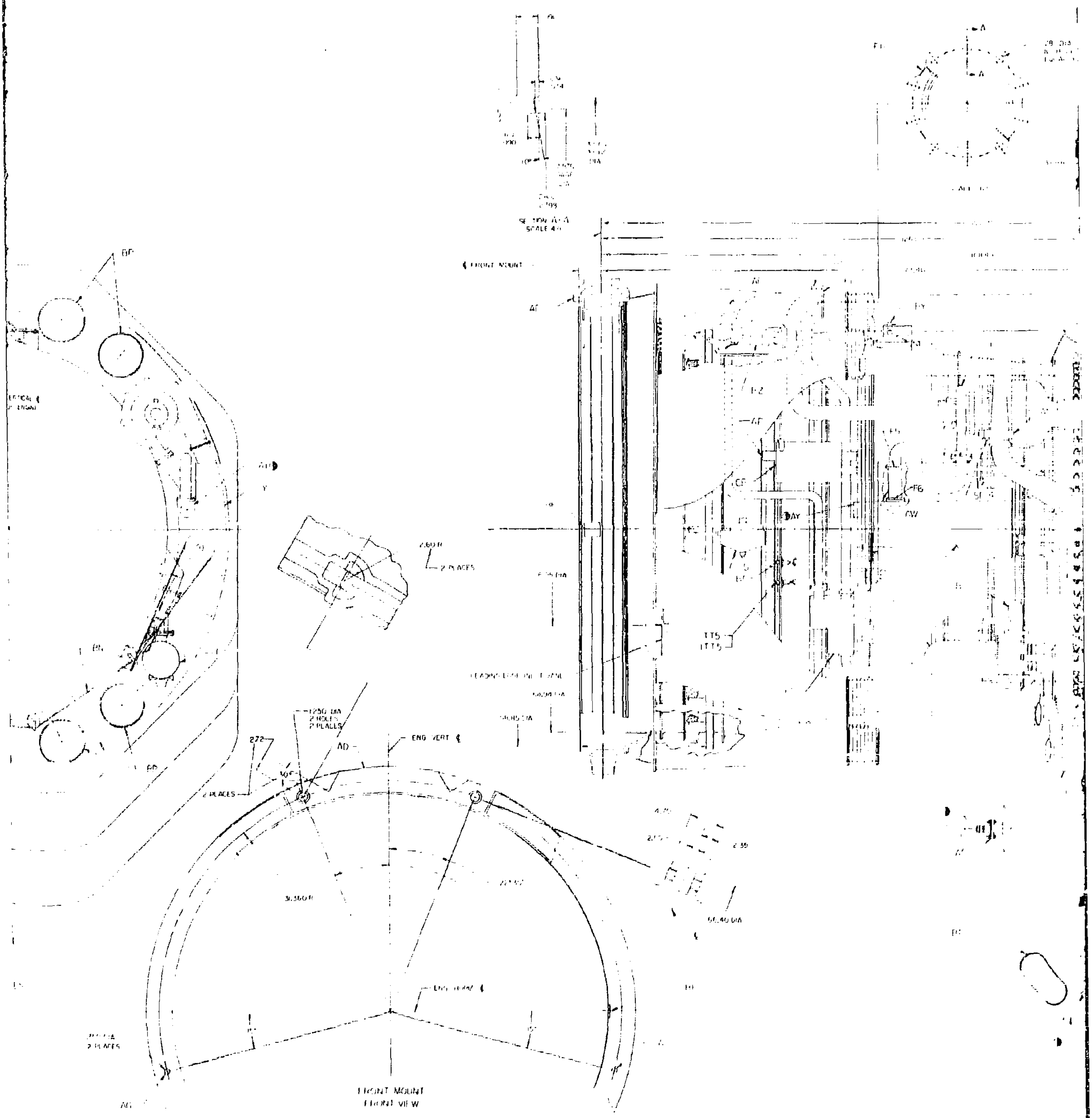
2





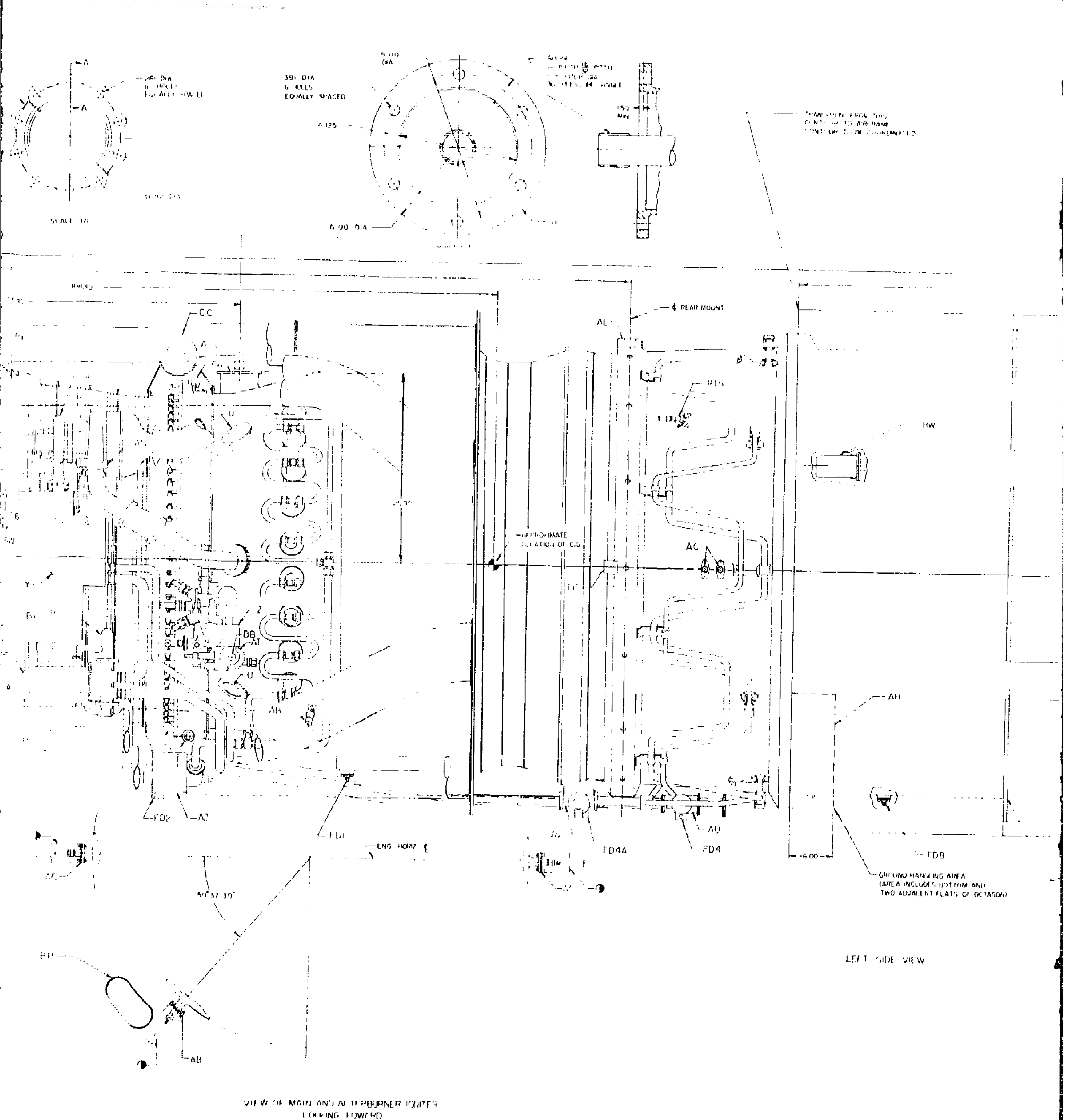
FRONT VIEW
SHOWING ACCESSORIES

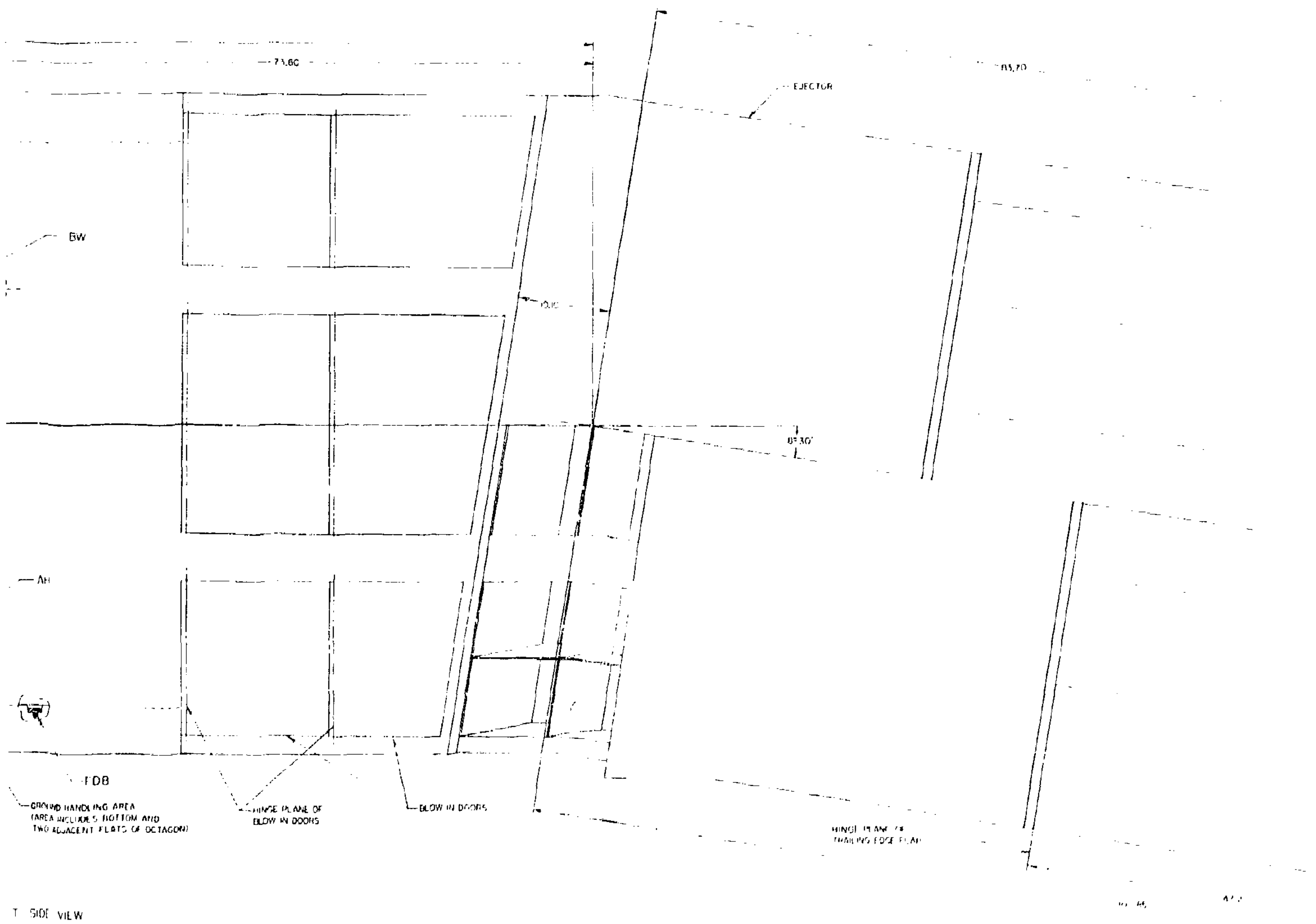
3



SECTION A-A
SCALE 4/1
FRONT MOUNT
AT
2.60 R
2 PLACES
LEADING EDGE INLET VANE
.1250 DIA
2 HOLES
2 PLACES
AD
ENG VERT ℄
272
2 PLACES
30.360 R
66.40 DIA
2.35
ENG HORIZ ℄
2 PLACES
FRONT MOUNT
FRONT VIEW
AG
BP
AY
AW
TT5
TT5
F6


4


SCALE 1/1
6.00 DIA
REAR MOUNT
APPROXIMATE
LOCATION OF C.G.
CC
AC
AH
BB
AB
FD2
FD1
FD4A
FD4
FD8
AU
ENG HORZ ℄
6.00
GROUND HANDLING AREA
(AREA INCLUDES BOTTOM AND
TWO ADJACENT FLATS OF OCTAGON)
LEFT SIDE VIEW
VIEW OF MAIN AND AFTERBURNER IGNITER
LOOKING FOWARD
5


73.60
83.70
EJECTOR
BW
AH
FDB
GROUND HANDLING AREA
(AREA INCLUDES BOTTOM AND
TWO ADJACENT FLATS OF OCTAGON)
HINGE PLANE OF
BLOW IN DOORS
BLOW IN DOORS
HINGE PLANE OF
TRAILING EDGE FLAP

T SIDE VIEW

TORQUE & MOMENT LIMITATION TABLE-ACCESS DRIVES

| DRIVE PAD | NOMINAL USE | TORQUE (LB. IN.) CONTINUOUS | OVERLOAD | STATIC | SPEED RATIO | ROTATION (FACING ENGINE PAD) | OVERHUNG MOMENT (LB. IN.) |
|---|---|---|---|---|---|---|---|
| A | POWER TAKE OFF | 2334 | 3500 [ ] | 24210 ® | 2,940 N | CCW | |
| B | TACH. | 7 | | 50 | .762 N | CCW | |
| C | HYD PUMP | 946 | 1420 | 4750 | .910 N | CCW | |

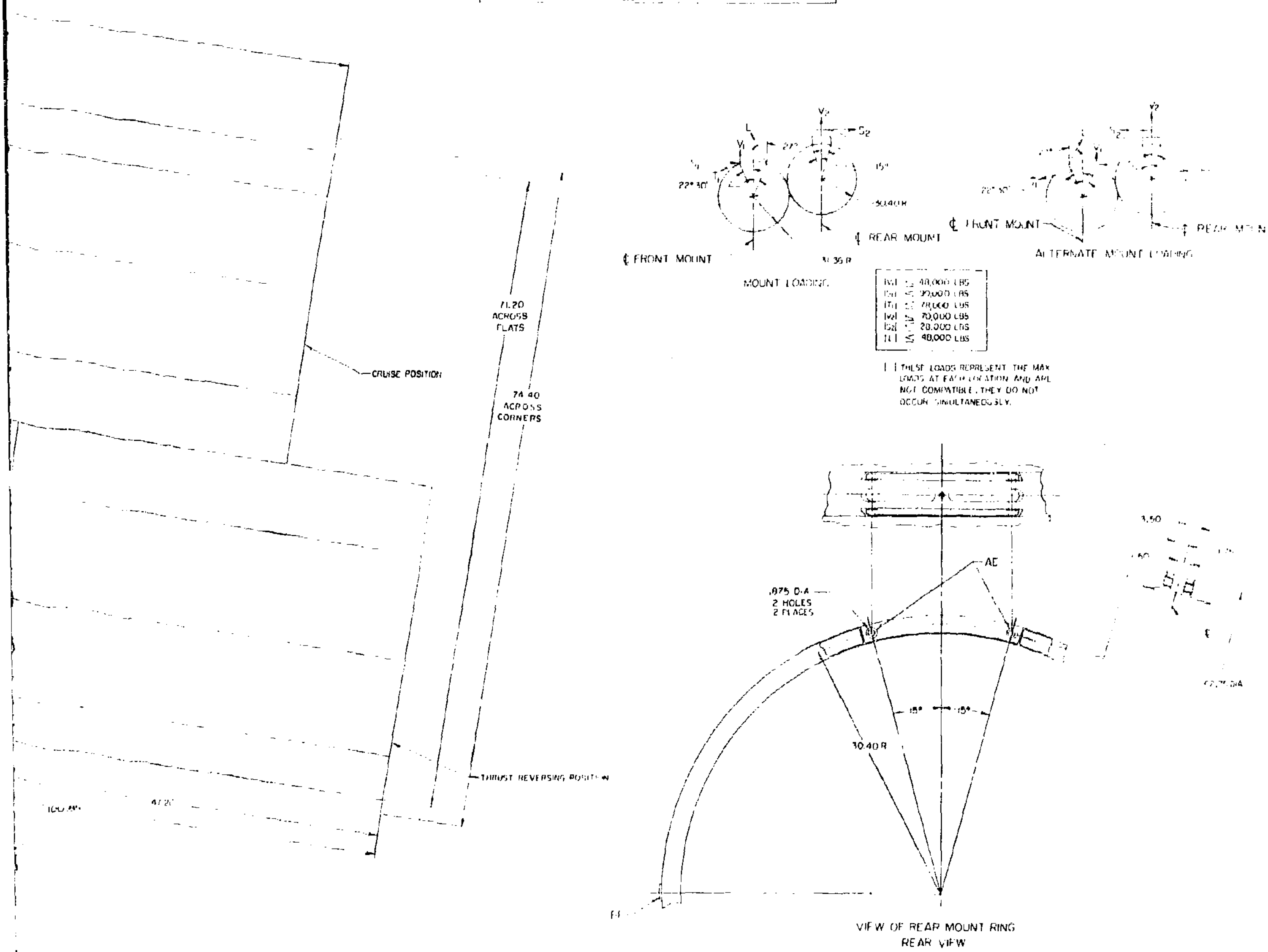
MAX ALLOWABLE CONTINUOUS TORQUE VALUES ARE AT ANY ENGINE SPEED UNLESS OTHERWISE SHOWN PROVIDED NO DESTRUCTIVE FORCES RESULTING FROM ACCESS TORSIONAL VIBRATION ARE PRESENT

MAX ALLOWABLE OVERHUNG BENDING MOMENTS OF ACCESSORIES ABOUT DRIVE PAD FACE ARE AS SHOWN, PROVIDED NO DESTRUCTIVE FORCES RESULTING FROM ACCESS VIBRATION ARE PRESENT

[ ] — MAX ALLOWABLE FOR 5 MINUTE DURATION RECURRING AT 4 HR INTERVALS MIN

® — P.T.O. DRIVE SHALL HAVE ADEQUATE STRENGTH TO ACCOMMODATE A MAX TORQUE EQUIV TO 92% [illegible] LB STARTER DRIVE. SHEAR SECTION SHALL FAIL AT A STATIC TORQUE EQUIV TO [illegible] LB OR [illegible] FT LB AT P.T.O. PAD

N REFERS TO ENGINE ROTOR SPEED




STJ227B ENGINE INSTALLATION (BO

Figure 2B-91

| TORQUE & MOMENT LIMITATION TABLE-ACCESS DRIVES | | | | | | | |
|---|---|---|---|---|---|---|---|
| DRIVE PAD | NOMINAL USE | TORQUE (LB IN) | | | SPEED RATIO | ROTATION (FACING ENGINE PAD) | OVERHUNG MOMENT (LB IN) |
| | | CONTINUOUS | OVERLOAD | STATIC | | | |
| A | POWER TAKE OFF | 2374 | [illegible] | [illegible] | 2.940 N | CCW | [illegible] |
| B | TACH | 7 | | 50 | .702 N | CCW | — |
| C | [illegible] | [illegible] | 1420 | 4730 | .990 N | CCW | — |

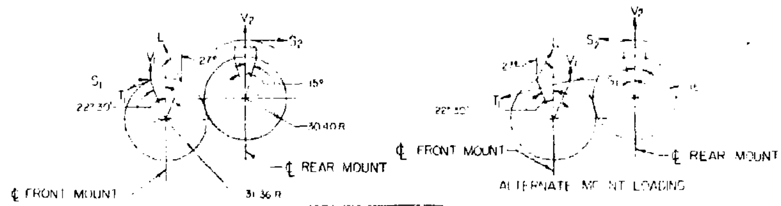
MAX ALLOWABLE CONTINUOUS TORQUE VALUES ARE AT ANY ENGINE SPEED UNLESS OTHERWISE SHOWN PROVIDED NO DESTRUCTIVE FORCES RESULTING FROM ACCESS TORSIONAL VIBRATION ARE PRESENT

MAX ALLOWABLE OVERHUNG BENDING MOMENTS OF ACCESSORIES ABOUT DRIVE PAD FACE ARE AS SHOWN PROVIDED NO DESTRUCTIVE FORCES RESULTING FROM ACCESS VIBRATION ARE PRESENT

MAX ALLOWABLE FOR 5 MINUTE DURATION RECURRING AT [illegible] INTERVALS MIN

P.T.O. DRIVE SHALL HAVE ADEQUATE STRENGTH TO ACCOMODATE A MAX TORQUE EQUAL TO 92[illegible] FT LB STARTER DRIVE SHEAR SECTION SHALL FAIL AT A STATIC TORQUE EQUAL TO [illegible] FT LB AT P.T.O. PAD

N REFERS TO ENGINE [illegible] SPEED

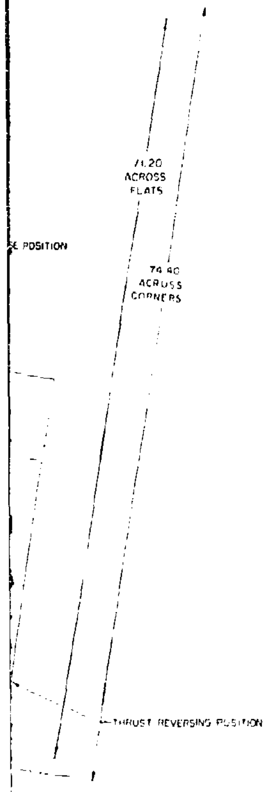


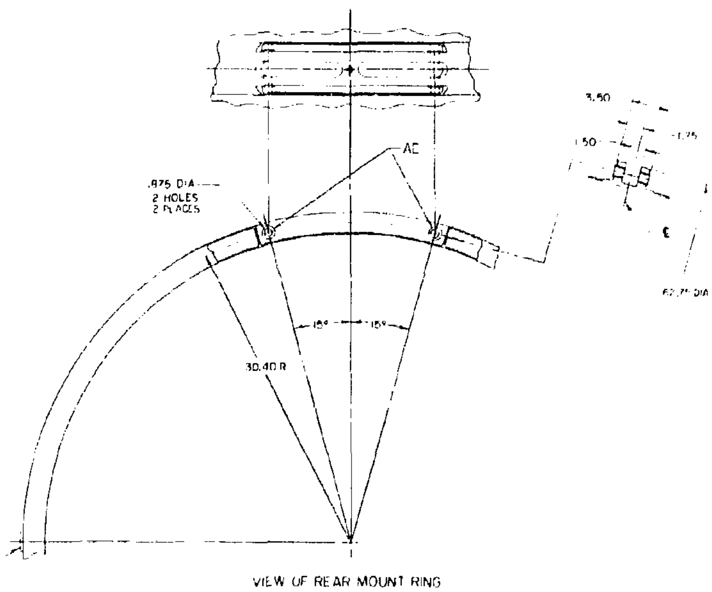

MOUNT LOADING

ALTERNATE MOUNT LOADING

| | | |
|---|---|---|
| $\lvert V_1 \rvert$ | $\leq$ | 48,000 LBS |
| $\lvert S_1 \rvert$ | $\leq$ | 90,000 LBS |
| $\lvert T_1 \rvert$ | $\leq$ | 78,000 LBS |
| $\lvert V_2 \rvert$ | $\leq$ | 70,000 LBS |
| $\lvert S_2 \rvert$ | $\leq$ | 28,000 LBS |
| $\lvert L \rvert$ | $\leq$ | 48,000 LBS |

| | THESE LOADS REPRESENT THE MAX LOADS AT EACH LOCATION AND ARE NOT COMPATIBLE. THEY DO NOT OCCUR SIMULTANEOUSLY.




VIEW OF REAR MOUNT RING
REAR VIEW

PRELIMINARY INSTALLATION DWG
PRINTED JUNE 16,1965

④— MIN SPACE FOR REMOVAL

⑤ DATUMS FOR TRUE POSITION OF FEATURES SHOWN ARE CENTERLINES OF CASE CONTAINING SUCH FEATURES UNLESS OTHERWISE INDICATED

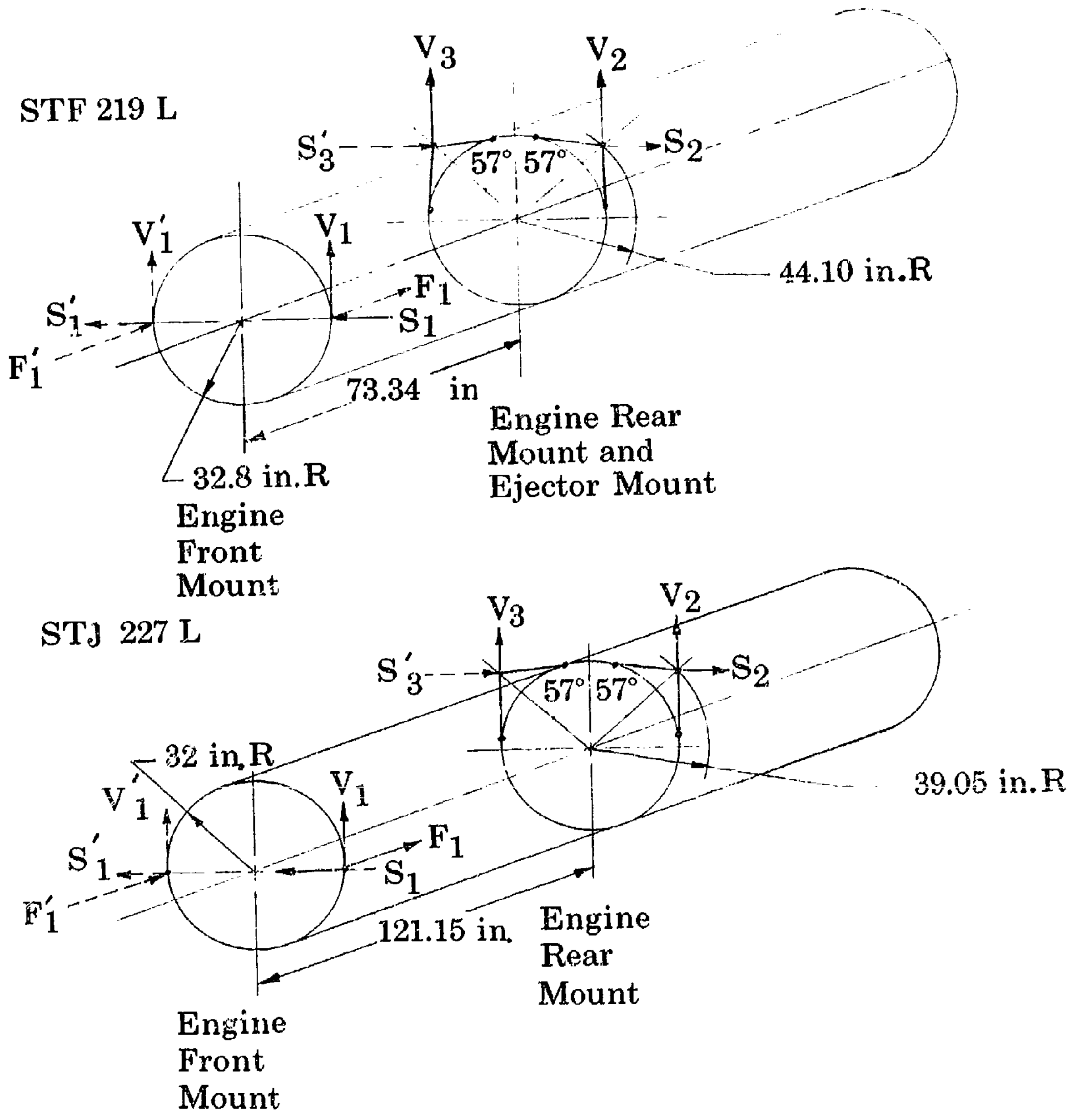
②— ALL DIMENSIONS SHOWN ARE NOMINAL UNLESS OTHERWISE SHOWN

①— ALL DIMENSIONS GIVEN FOR ROOM TEMPERATURE

8

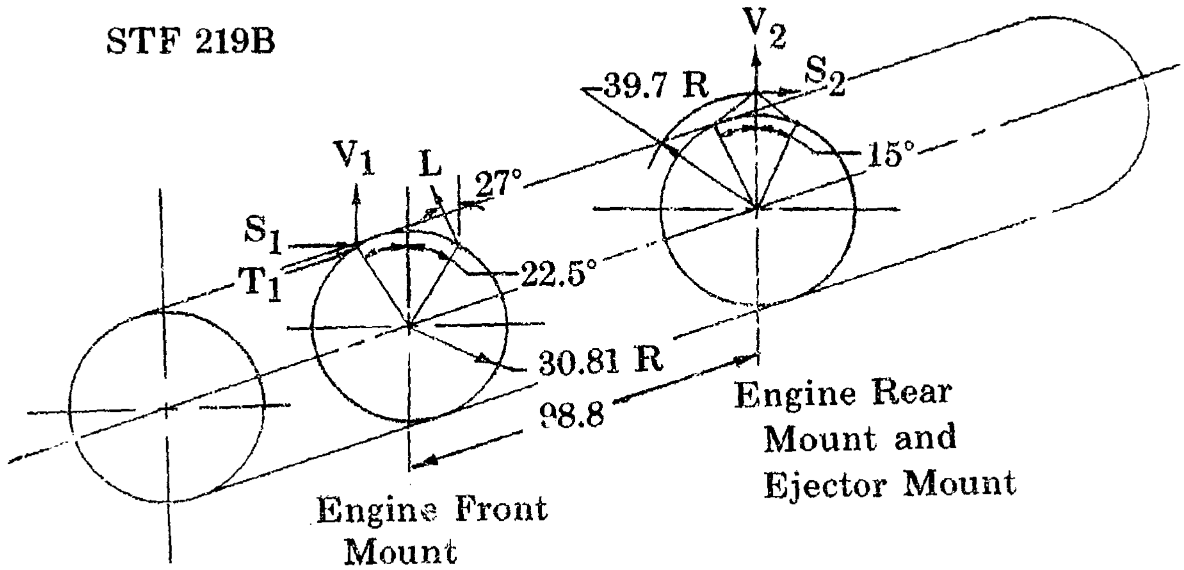
STJ227B ENGINE INSTALLATION (BOEING)

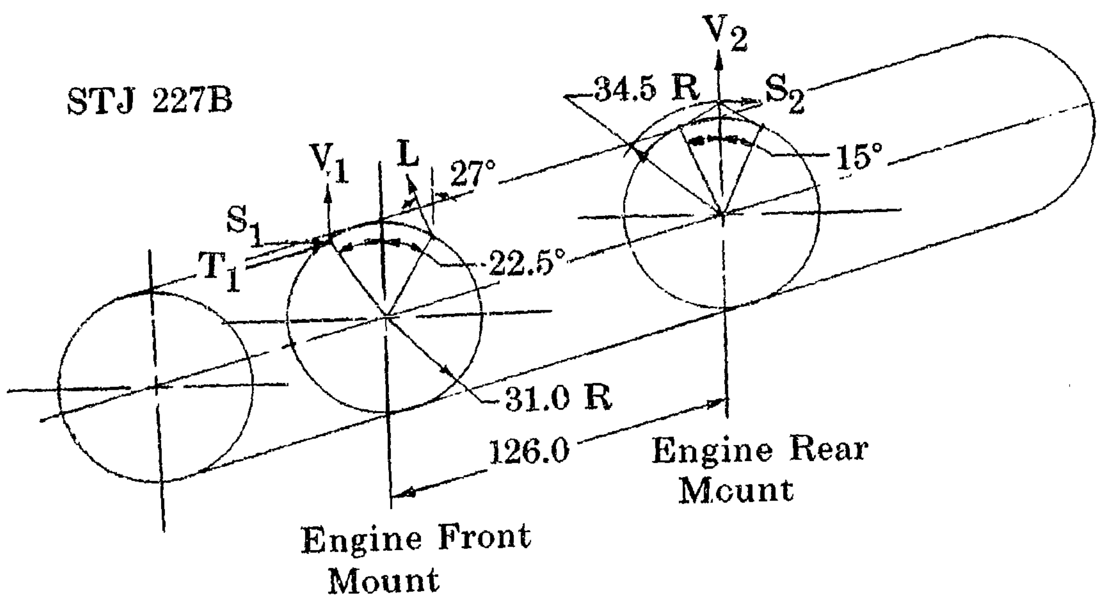
Figure 2B-91




ENGINE MOUNTING SYSTEM (LOCKHEED)

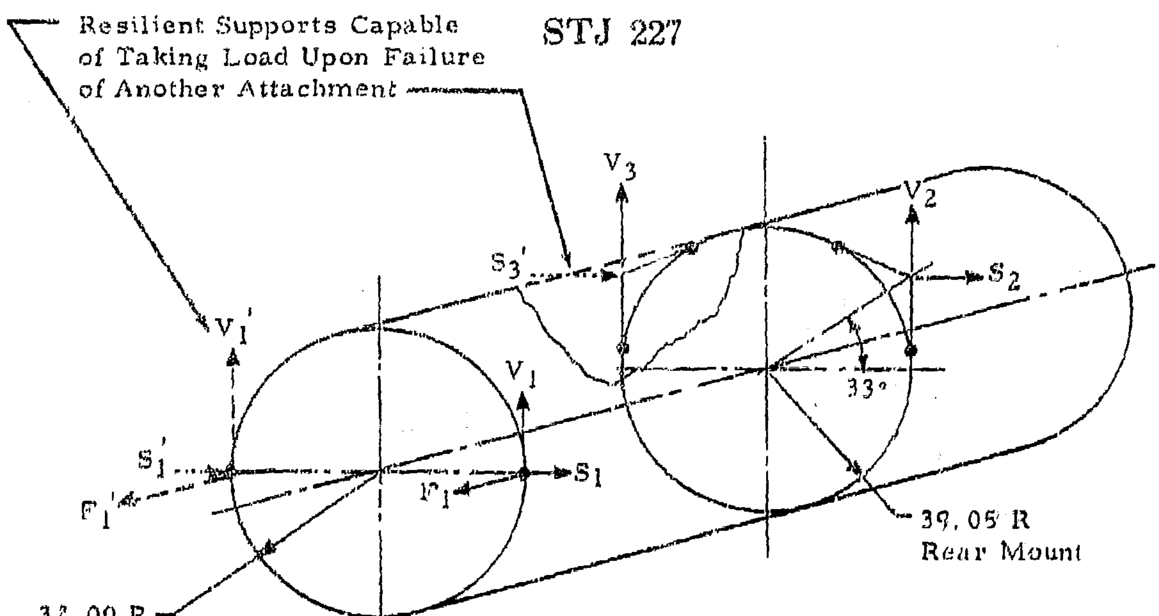
Figure 2B-92




ENGINE MOUNTING SYSTEM (BOEING)

Figure 2B-93



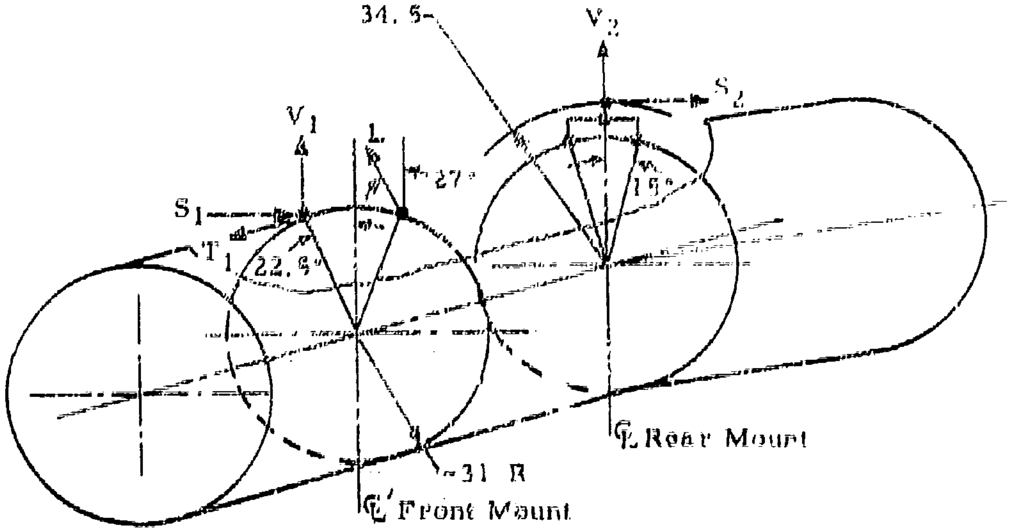

| | |
|---|---|
| Distance Between Mount Planes | 121.15 in. |
| Engine Weight | 10,585 lb |
| Engine C. G. From Front Mount | 104.2 in. Aft |

The Allowable Limit Loads for Each Mount Attachment Point Must Satisfy all Equations Below:

| | |
|---|---|
| $V_1 \leq 10,000$ lb | $V_1' \leq 10,000$ lb |
| $S_1 \leq 25,000$ lb | $S_1' \leq 25,000$ lb |
| $F_1 \leq 80,000$ lb | $F_1' \leq 80,000$ lb |
| $V_2 \leq 35,000$ lb | $S_3' \leq 40,000$ lb |
| $S_2 \leq 40,000$ lb | |
| $V_3 \leq 40,000$ lb | |

MAXIMUM FLIGHT LOADS (LOCKHEED)

Figure 2B-94




| | |
|---|---|
| Distance Between Mount Planes | 126 in. |
| Engine Weight (Including Inlet) | 12,495 lb |
| Engine C.G. From Front Mount | 110 in. Aft |

The Allowable Limit Loads for Each Mount Attachment Point Must Satisfy all Equations Below:

$$V_1 \le 48,000 \text{ lb}$$

$$S_1 \le 90,000 \text{ lb}$$

$$T_1 \le 78,000 \text{ lb}$$

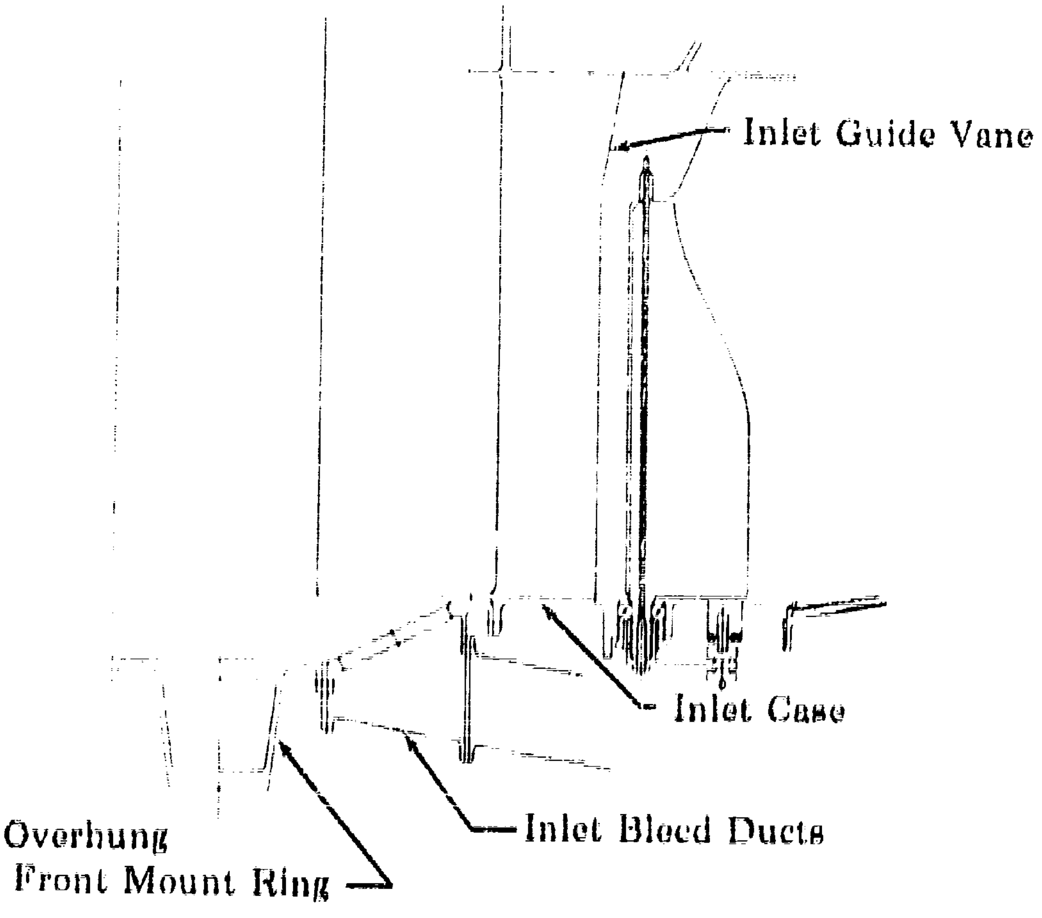
$$V_2 \le 70,000 \text{ lb}$$

$$S_2 \le 28,000 \text{ lb}$$

$$L_1 \le 48,000 \text{ lb}$$

MAXIMUM FLIGHT LOADS (BOEING)

Figure 2B-95




OVERHUNG FRONT MOUNT SYSTEM (BOEING)

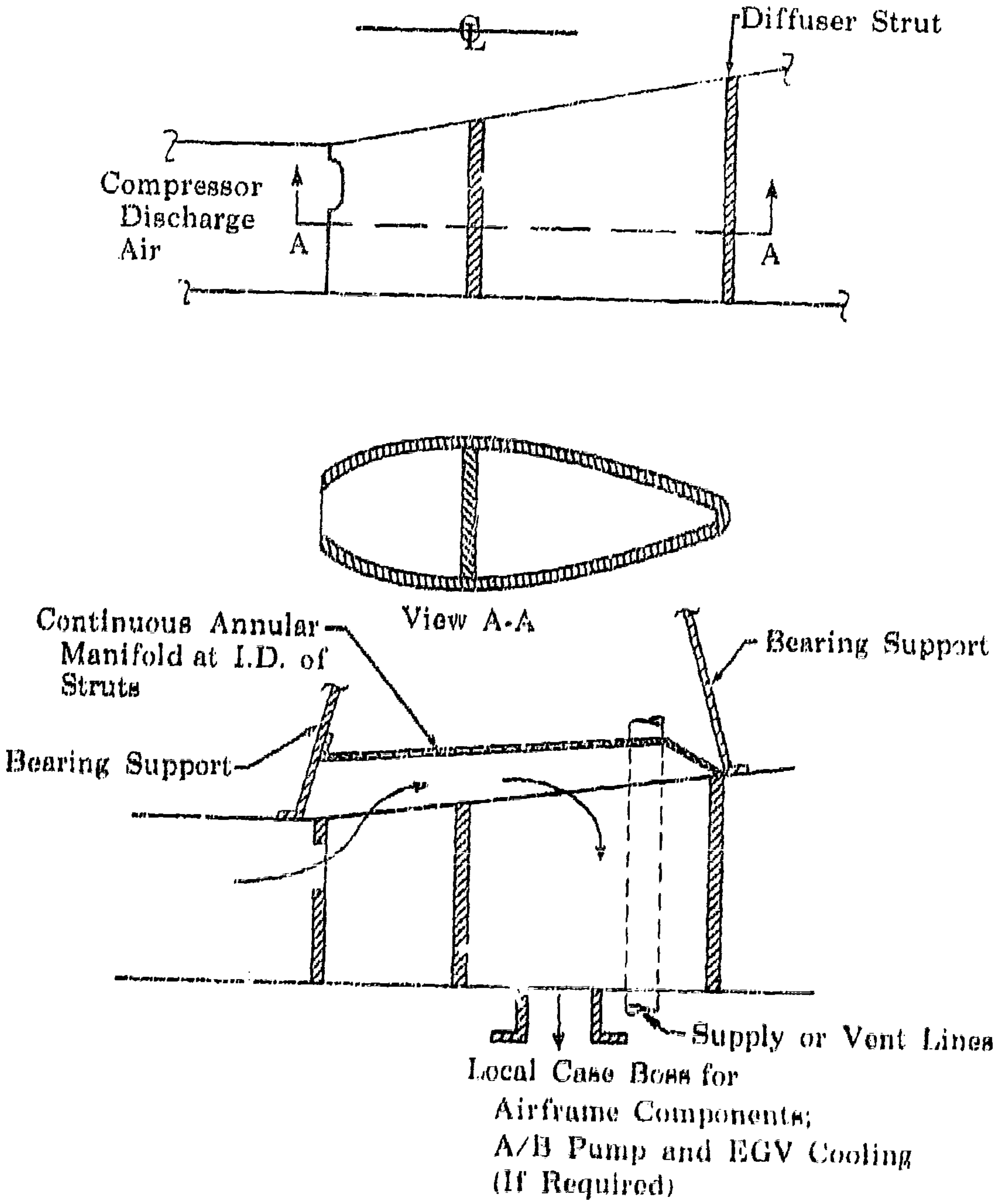
Figure 2B-96

| Bleed Usage | Bleed Source | SLTO | % Engine Airflow Accel | (Mn = 2.5, 65K) Cruise | Remarks |
|---|---|---|---|---|---|
| Starting | Fourth Stage Stator | Not Defined | | | |
| Seal Pressurization | Fourth Stator Exit | 0.3 | 0.2 | 0.1 | Strut area required for supply and dump tubes |
| [illegible]e Cabin | Diffuser Struts | 0.64 | 0.51 | 0.74 | |
| Airframe Fuel De-Icing | Diffuser Struts | 0.56 | | | |
| Airframe Anti-Icing | Diffuser Struts | 0.45 | 0.45 | | |
| Engine Inlet Guide Vane Anti-Icing | Diffuser Struts | 1.0 | 1.0 | | |
| Engine Afterburner Turbopump Actuation | Diffuser Struts | 0.94 | 0.80 | 0.56 | |
| Engine Exit Guide Vane Cooling | Rear of Last Compressor Disk | 0.75 | 0.75 | 0.75 | Strut area supplied to dump chamber to ambient |
| Afterburner Duct | Diffuser Struts | 2.0 | 2.0 | 2.0 | Required for 2300°F $T_{t4}$ only |
| Exit Guide Vane Cooling | Diffuser Struts | 0.5 | 0.4 | 0.3 | Required for 2300°F $T_{t4}$ only |

STJ227 BLEED SYSTEM REQUIREMENTS

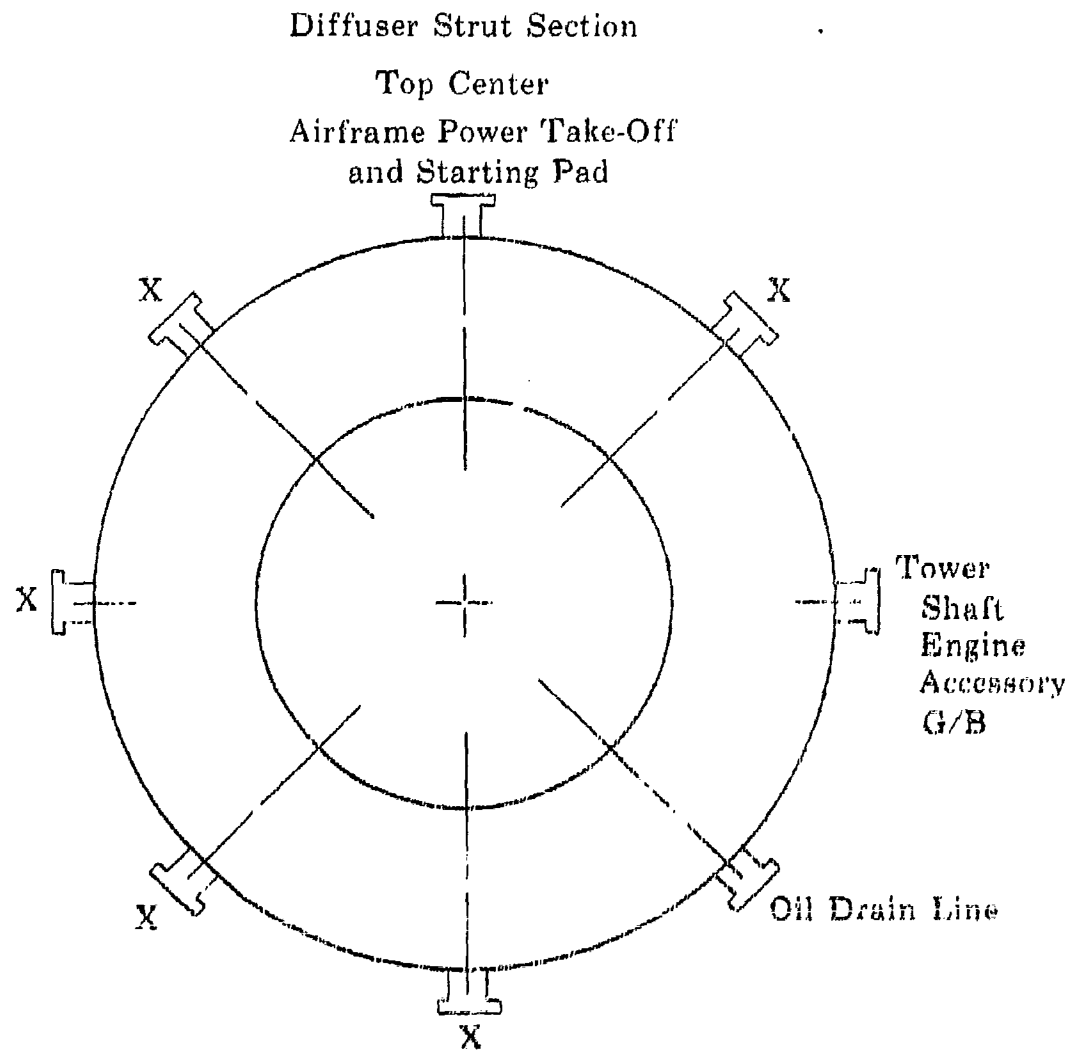
Figure 2B-97

Design Considerations




AIRFRAME BLEED FLOW REQUIREMENTS

Figure 2B-98



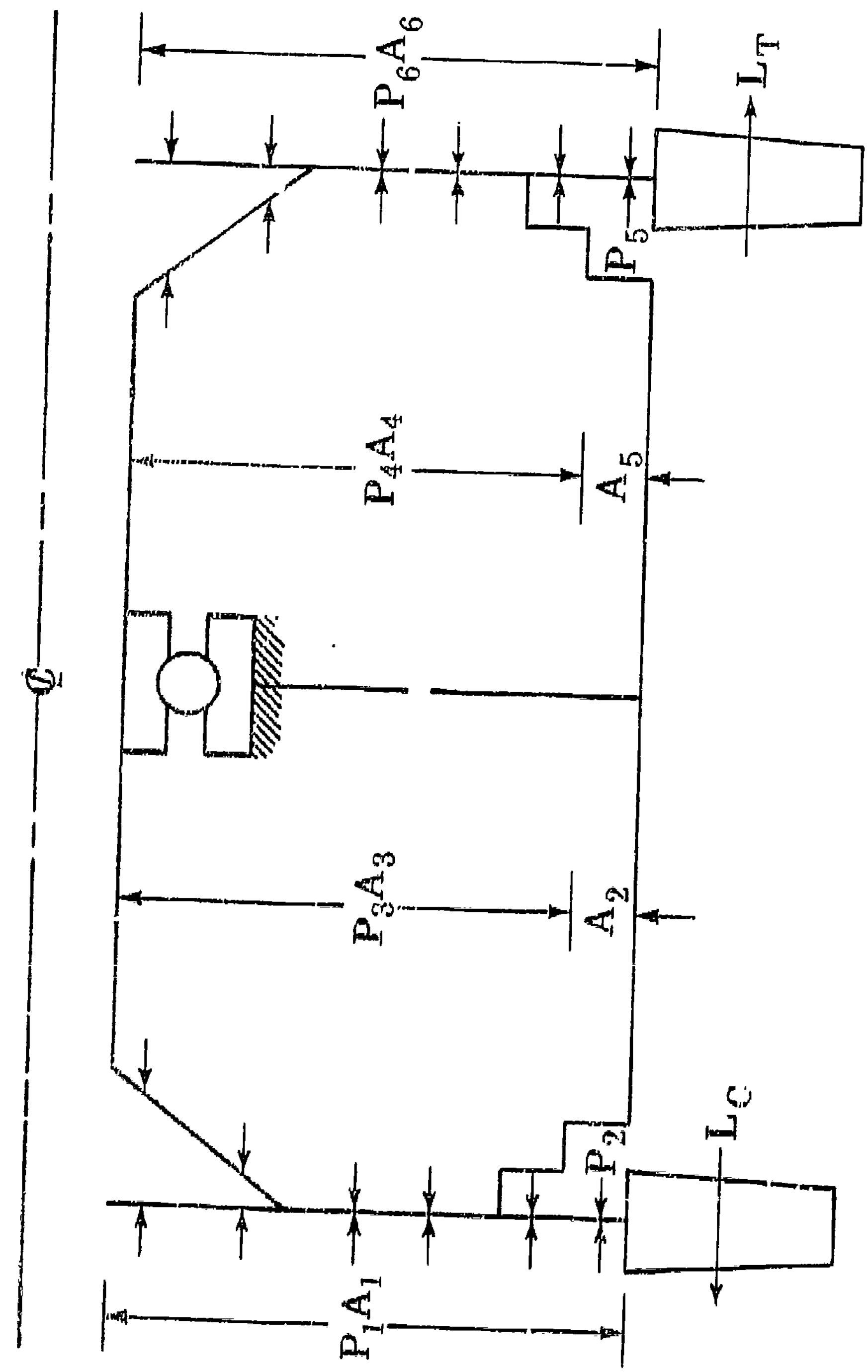

Total Available Area/Strut - 13 in.$^2$

Note: 1 - X Represents Available Struts for Supply or Vent Areas

2 - All Struts Will Have Leading Edge Slots

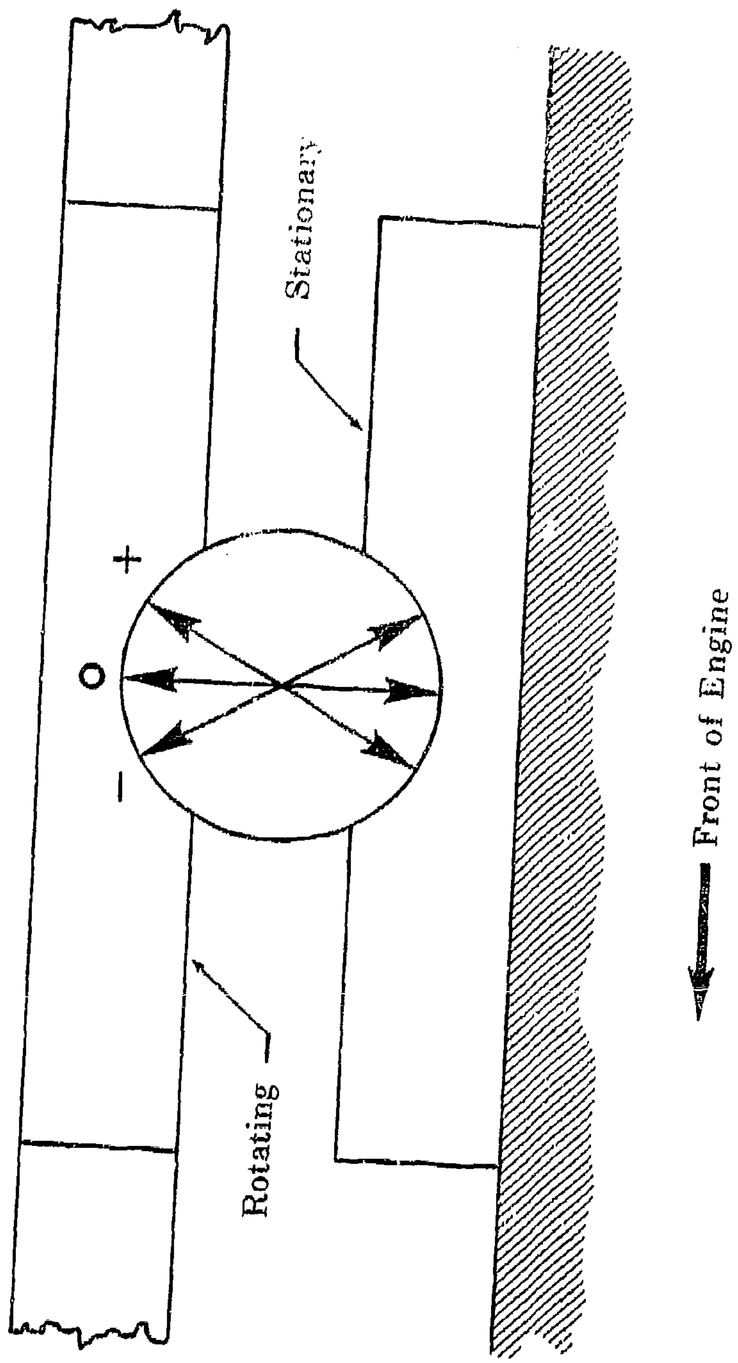
ENGINE BLEED FLOW REQUIREMENTS

Figure 2B-99



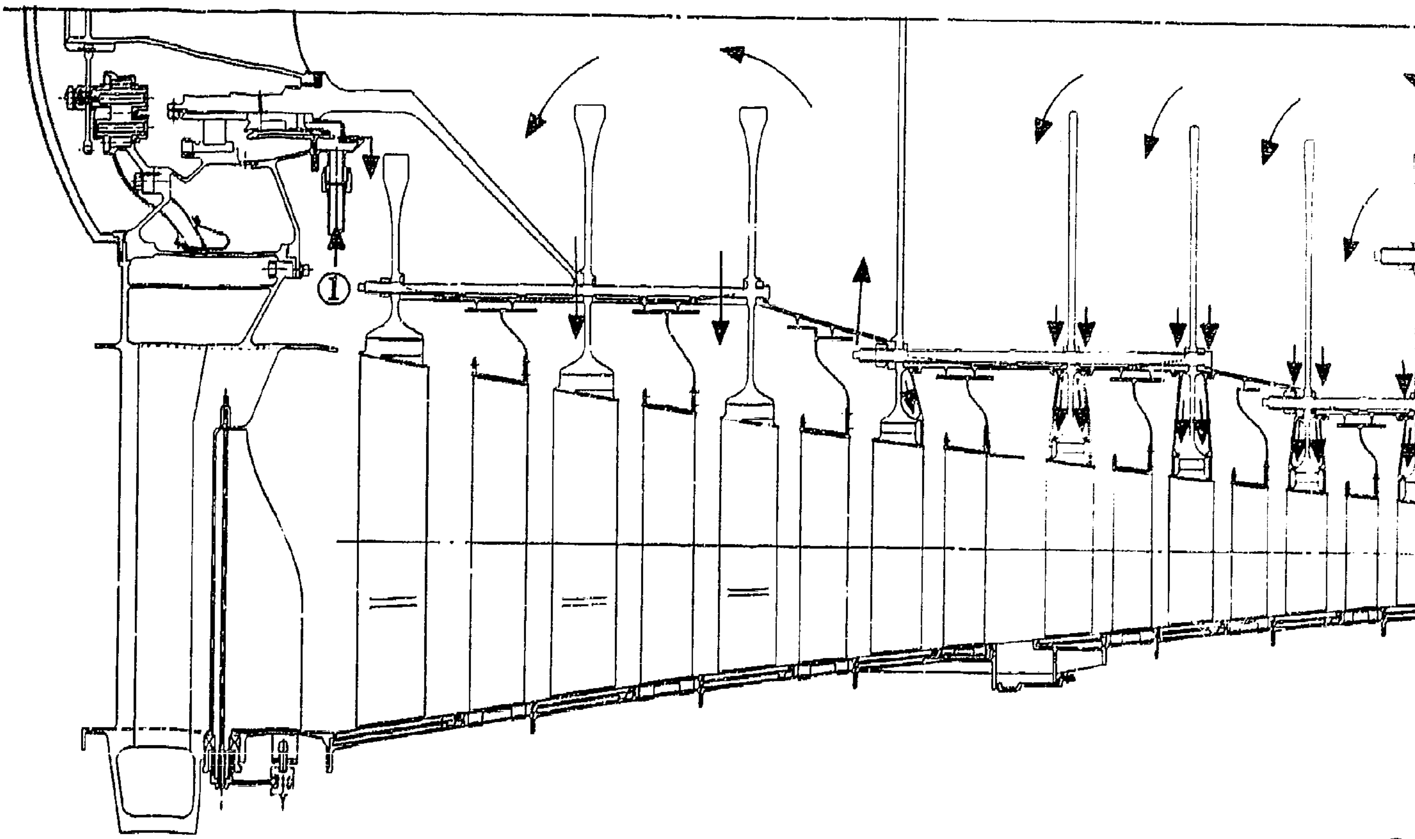

TYPICAL FORCES ACTING ON A TURBOJET ROTOR

Figure 2B-100



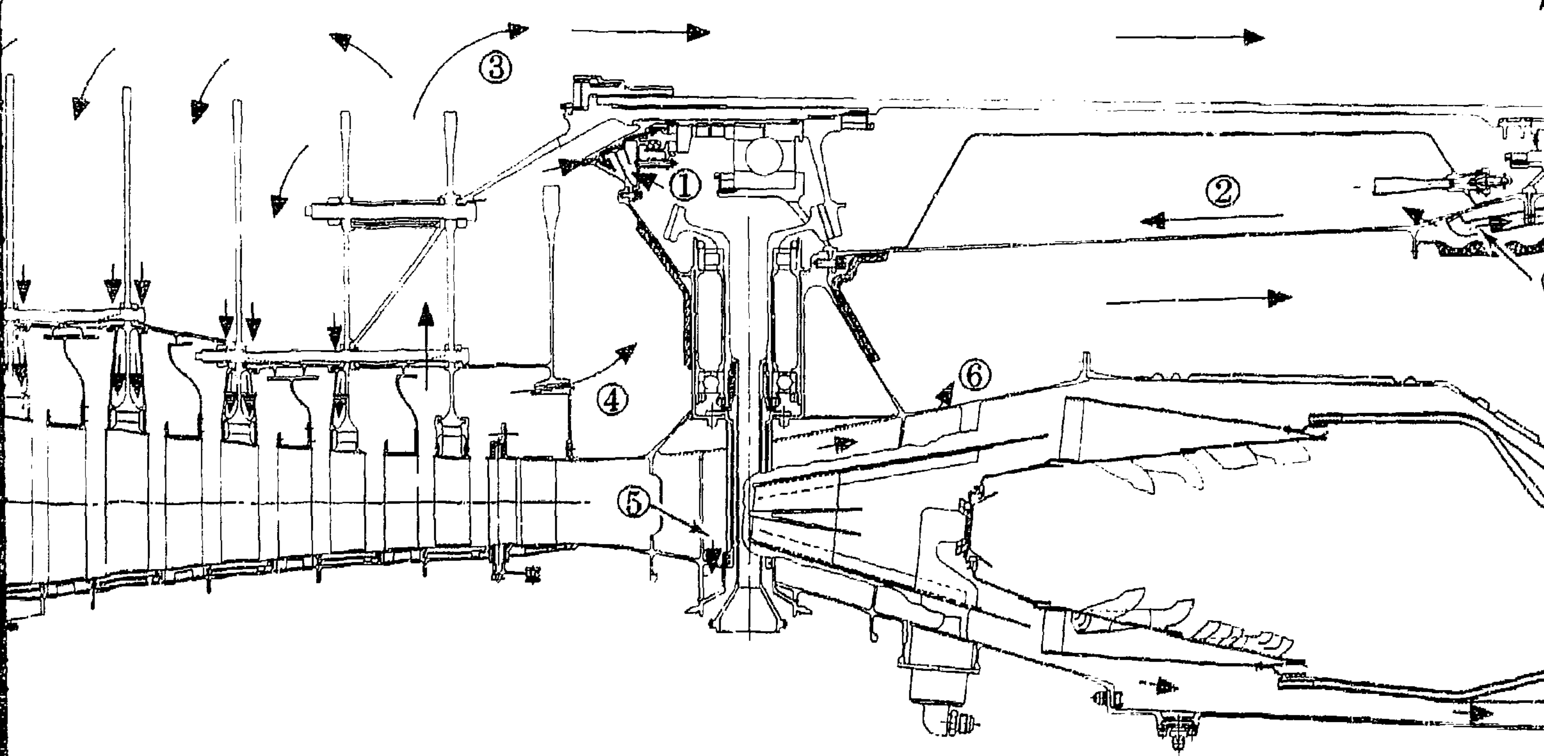

LOAD ACTION ON A BEARING

Figure 2B-101




①
②
③
④
⑤
⑥

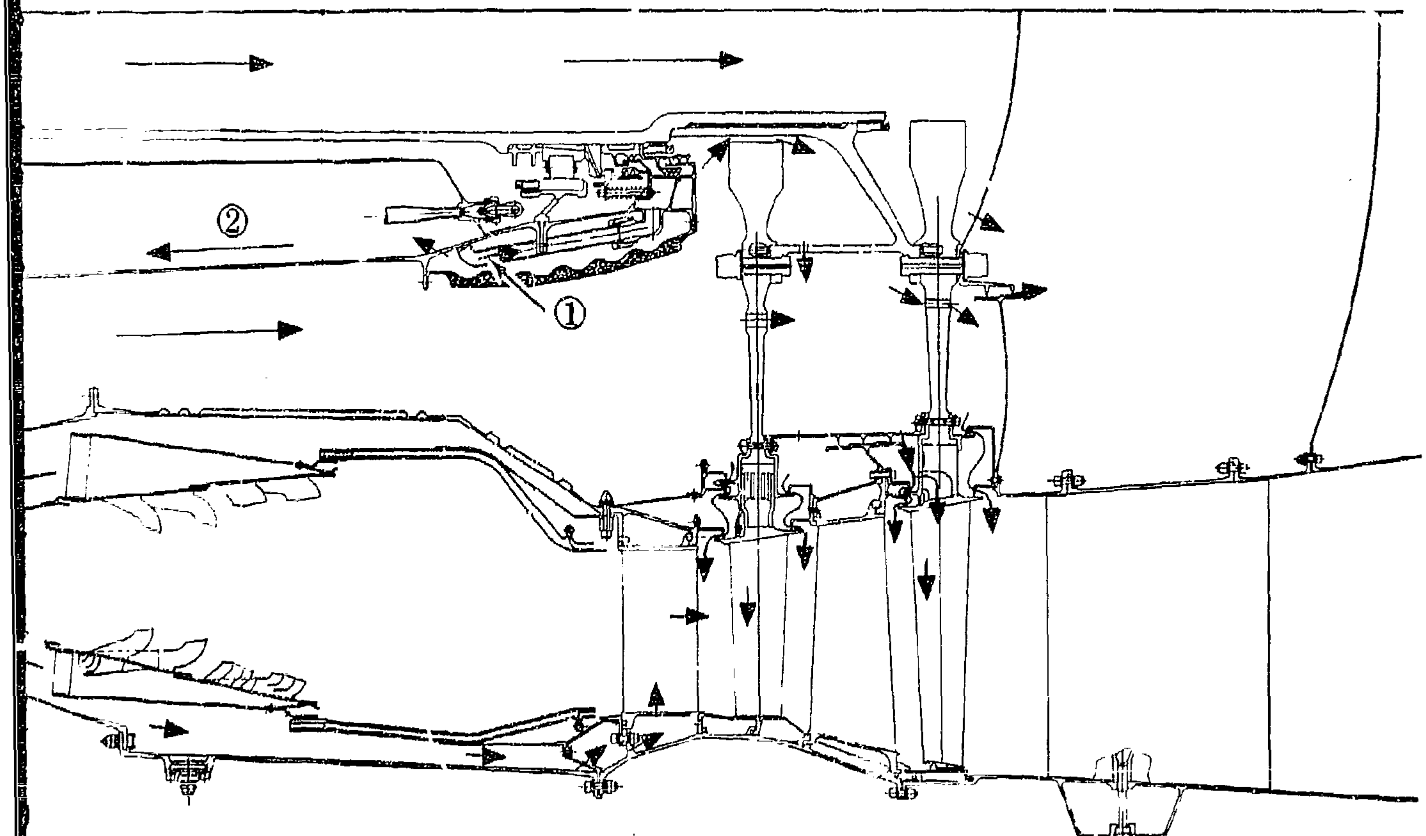
1




① No. 4 Compressor Stator Supply Air
② No. 3 Seal Dump Flow
③ No. 8 Stator Cooling Flow
④ No. 9 Labyrinth Seal Flow
⑤ To Ambient
⑥ Inner Burner Case Flow

STJ227

2




STJ227 FLOW SCHEMATIC FOR ENGINE THRUST
BALANCE AND COOLING

Figure 2B-102

3

(Jet Pump Not Included)

| | Q Btu/min | $\dot{w}$ lb/min | Scoop Efficiency % | $\dot{w}'$ lb/min | Excess Oil $\dot{w}' - \dot{w}$ |
|---|---|---|---|---|---|
| Number 1 Bearing Compartment | | | | | |
| Seals | 250 | 8.3 | | 8.3 | |
| Roller Bearing | 100 | 3.3 | | 3.3 | |
| Amb. & Conduction | 77 | 2.6 | | 2.6 | |
| Total | 427 | | | 14.2 | |
| Number 2 Bearing Compartment | | | | | |
| Carbon Seal | 296 | 9.9 | 70 | 14.1 | 4.2 |
| Ball Bearing (1) | 740 | 24.6 | 70 | 35.2 | 10.6 |
| A. C. Gear | 255 | 8.5 | | 8.5 | |
| M. E. Gear | 130 | 4.3 | | 4.3 | |
| A.C.T.S. Ball Brg. | 25 | 0.8 | | 0.8 | |
| A.C.T.S. Roller Brg. | 50 | 1.6 | | 1.6 | |
| M.E.T.S. Ball Brg. | 25 | 0.8 | | 0.8 | |
| M.E.T.S. Roller Brg. | 50 | 1.6 | | 1.6 | |
| Amb. & Conduction | 167 | 5.6 | | | |
| Towershaft Cond. | 30 | 1.0 | | | |
| Total | 1768 | | | 66.9 | 14.8 |
| Number 3 Bearing Compartment | | | | | |
| Carbon Seal | 277 | 9.3 | 70 | 13.3 | 4.0 |
| Roller Bearing | 300 | 10.0 | 70 | 14.3 | 4.3 |
| Amb. & Conduction | 394 | 13.1 | | | |
| Total | 971 | | | 27.6 | 8.3 |
| Main Gearbox | 751 | 25.0 | | 25.0 | |
| Right Angle Gearbox | 300 | 10.0 | | 10.6 | |
| Oil Tank Ambient | 50 | 1.6 | | 1.6 | |

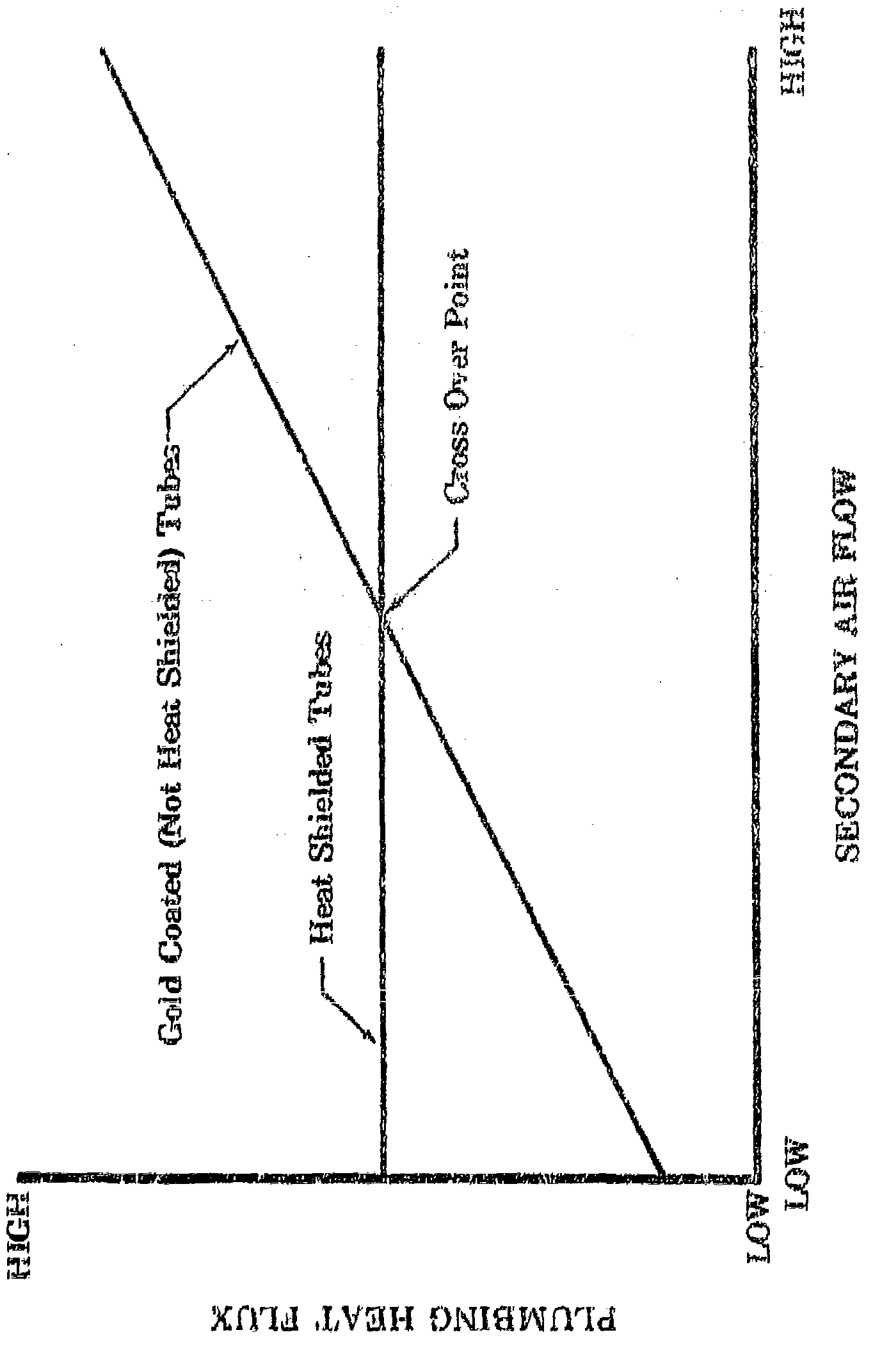
Total Flow = 14.2 + 66.9 + 27.6 + 25.0 + 10.6 + 1.6 = 145.9 lb/min + Jet Pump Flow

Total Q = 4267 Btu/min

Total 145.9 + 27.6 = 173.5 lb/min

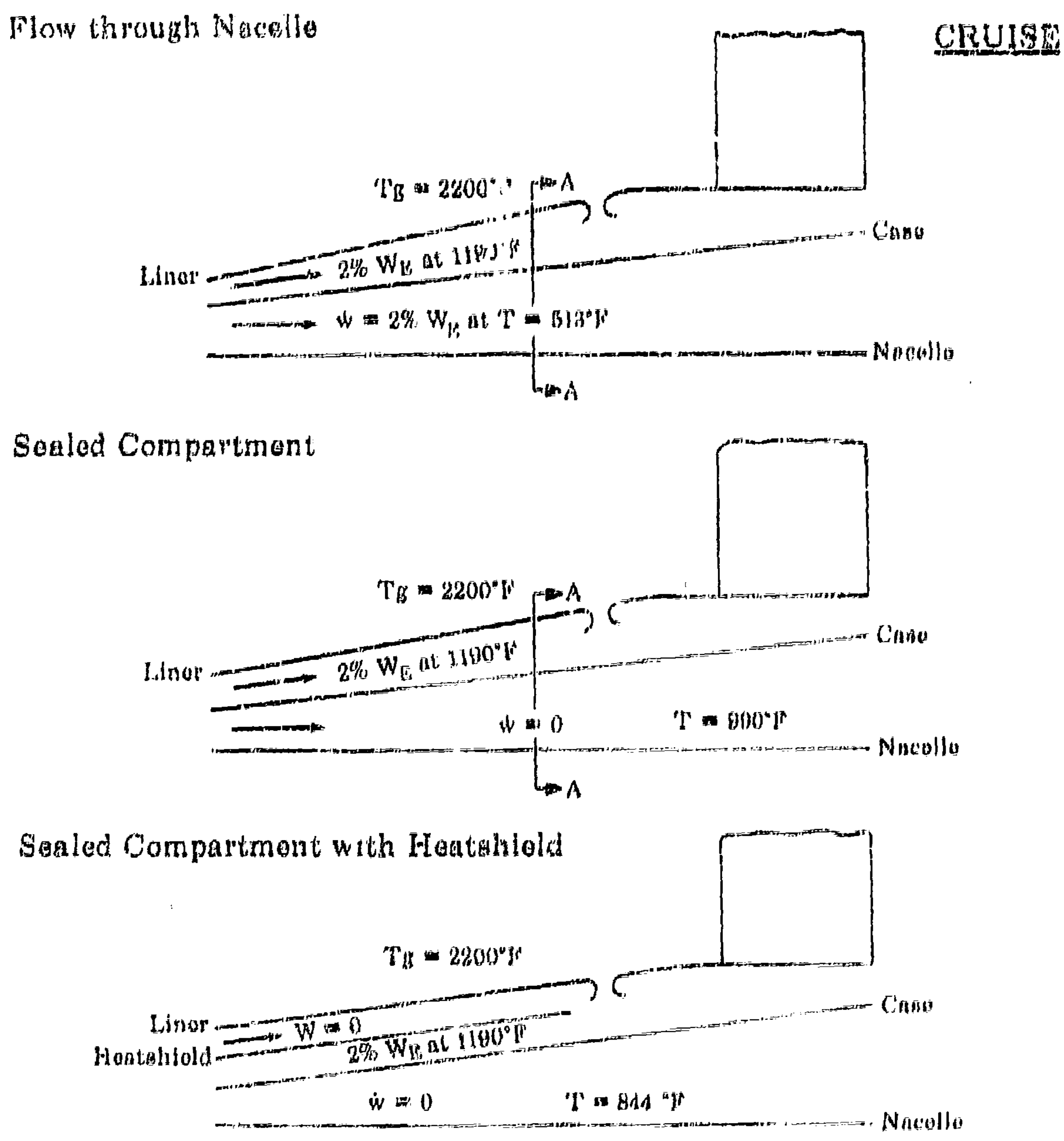
STJ227 OIL FLOWS

Figure 2B-103




CHARACTERISTIC HEAT FLUX TRENDS

Figure 2B-104




HEAT TRANSFER CALCULATIONS

Figure 2B-105

## CRUISE

| | | | |
|---|---|---|---|
| Flow through Nacelle | *513 °F | 1690 °F | Liner |
| | | 1380 °F | Case |
| | | 555 °F | Nacelle |
| Sealed Compartment | *990 °F | 1695 °F | Liner |
| | | 1415 °F | Case |
| | | 565 °F | Nacelle |
| Sealed Compartment with Heat Shield | *844 °F | 1750 °F | Liner |
| | | 1430 °F | Heat Shield |
| | | 1160 °F | Case |
| | | 530 °F | Nacelle |

*Air Temperatures

METAL TEMPERATURE HEAT TRANSFER RESULTS

Figure 2B-106

Air flow - 525 lb/sec
Flow schedule - high
Turbine Inlet temperature - 2000°F

| | |
|---|---|
| Compressor Inlet | 385 |
| Compressor Rotor | 1,010 |
| Compressor Stator | 750 |
| Diffuser | 650 |
| Burner | 875 |
| Turbine Rotor and Shafts | 1,070 |
| Turbine Stator | 810 |
| Turbine Exhaust | 650 |
| Afterburner | 1,235 |
| Ejector-Reverser | 2,000 |
| Accessories Drive | 185 |
| Components and Plumbing | 1,180 |
| TOTAL | 10,800 (lbs) |

STJ227 WEIGHT BREAKDOWN

Figure 2B-107

BOEING

| $W_{a2}$ (lb/sec) | Turbine Inlet Temp. (Max. °F) | Flow Schedule | Afterburner A/B | Afterburner Tailpipe (°F) Temperature | Ejector Equivalent Round Diameter (in.) | Ejector L/D | Net Weight (lb) | Cant (degrees) | Special Installation Features | Gross Weight (lb) |
|---|---|---|---|---|---|---|---|---|---|---|
| 525 | 2000 | High | Full | 3100 | 78 | 1.295 | 10,800 | 7° 55' | 570 | 11,370 |
| 525 | 2000 | Base | Full | 3010 | 75 | 1.295 | 10,470 | 8° 35' | 575 | 11,045 |
| 525 | 2000 | Low | Full | 3010 | 75* | 1.235 | 10,220 | 8° 50' | 580 | 10,790 |
| 525 | 2300 | High | Full | 3030 | 78 | 1.265 | 10,760 | 8° 9' | 570 | 11,330 |
| 525 | 2300 | Base | Full | 3200 | 75 | 1.265 | 10,455 | 8° 40' | 575 | 11,030 |
| 525 | 2300 | Low | Full | 3220 | 75* | 1.25 | 10,215 | 9° 40' | 585 | 10,800 |

LOCKHEED

| $W_{a2}$ (lb/sec) | Turbine Inlet Temp. (Max. °F) | Flow Schedule | Afterburner A/B | Afterburner Tailpipe (°F) Temperature | Ejector Equivalent Round Diameter (in.) | Ejector L/D | Net Weight (lb) | Cant (degrees) | Special Installation Features | Gross Weight (lb) |
|---|---|---|---|---|---|---|---|---|---|---|
| 525 | 2000 | High | Full | 3100 | 78 | 1.295 | 10,800 | 4° | 160 | 10,960 |
| 525 | 2000 | Base | Full | 3100 | 75 | 1.295 | 10,470 | 4° | 180 | 10,650 |
| 525 | 2000 | Low | Full | 3100 | 75* | 1.235 | 10,200 | 4° | 180 | 10,380 |
| 525 | 2300 | High | Full | 3010 | 78 | 1.265 | 10,760 | 4° | 180 | 10,940 |
| 525 | 2300 | Base | Full | 3010 | 75 | 1.265 | 10,455 | 4° | 180 | 10,635 |
| 525 | 2300 | Low | Full | 3010 | 75* | 1.25 | 10,215 | 4° | 180 | 10,395 |

*Optimum ejector diameters are less than the quoted values but current designs of actuating cylinders, linkages, secondary air flow passages and control protrusions prevent smaller ejector shroud and nacelle wrapping.

SPECIFIC INSTALLATION WEIGHT

Figure 2B-108